***Updated to Modification 36 ***

ATTACHMENT B

S-415-22

PERFORMANCE SPECIFICATION

FOR THE

GEOSTATIONARY OPERATIONAL ENVIRONMENTAL SATELLITE

GOES-N,O,P,Q

AUGUST 26, 1997

NASA/GODDARD SPACE FLIGHT CENTER
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ATTACHMENT B

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GEOSTATIONARY OPERATIONAL ENVIRONMENTAL SATELLITE

GOES-N,O,P,Q
PERFORMANCE SPECIFICATION

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GEOSTATIONARY OPERATIONAL ENVIRONMENTAL SATELLITE

GOES-N,O,P,Q

PERFORMANCE SPECIFICATION

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1.0 SCOPE

The mission of the GOES System is to acquire and disseminate environmental data from a near-equatorial Earth orbit at geostationary altitude. This includes making the measurements of the Earth's atmosphere, its surface, cloud cover and the solar and geosynchronous space environment. A major function of the GOES System is to support the Imager, Sounder, Solar X-ray Imager, and SEM instruments. Other functions of the GOES System are to: (1) support a collection of terrestrial and oceanographic Data Collection Platforms (DCPs), (2) relay Weather Facsimile (WEFAX) and imaging and sounding data between earth terminals, (3) relay the Emergency Managers Weather Information Network (EMWIN) broadcast, and (3) provide rapid detection of distress messages from Emergency Locator Transmitters (ELTs) and Emergency Position Indicating Radio Beacons (EPIRBs).

This GOES N-Q Performance Specification S-415-22 describes the performance requirements for the GOES N-Q Program. Separate ICDs describe other elements that interface with the GOES N-Q spacecraft.

Figure 1.0 depicts the primary interfacing elements of the GOES N-Q spacecraft and the ICD documents that describe the respective interfaces. System elements specified by this performance specification are:

1. The GOES N-Q spacecraft bus and the Space Environment Monitor (SEM) instruments.
2. Interfaces for the GFE instruments and associated GSE.
3. The GOES N-Q Spacecraft Support Ground System.
4. Flight operations hardware, software, and support.
5. Launch vehicle and launch services.

Interfaces to external users (DCS, SAR, and GVAR are to remain compatible with the existing GOES I-M system. The WEFAX is being upgraded from the current analog format to a digital standard developed by the Europeans, and the EMWIN will evolve from the present, limited demonstration project to a fully operational system. The existing ground system, including the ground portion of image navigation and registration (INR), is available for use by the GOES N-Q spacecraft contractor. Performance estimates, analysis, and any changes to the ground system are the responsibility of the GOES spacecraft contractor.

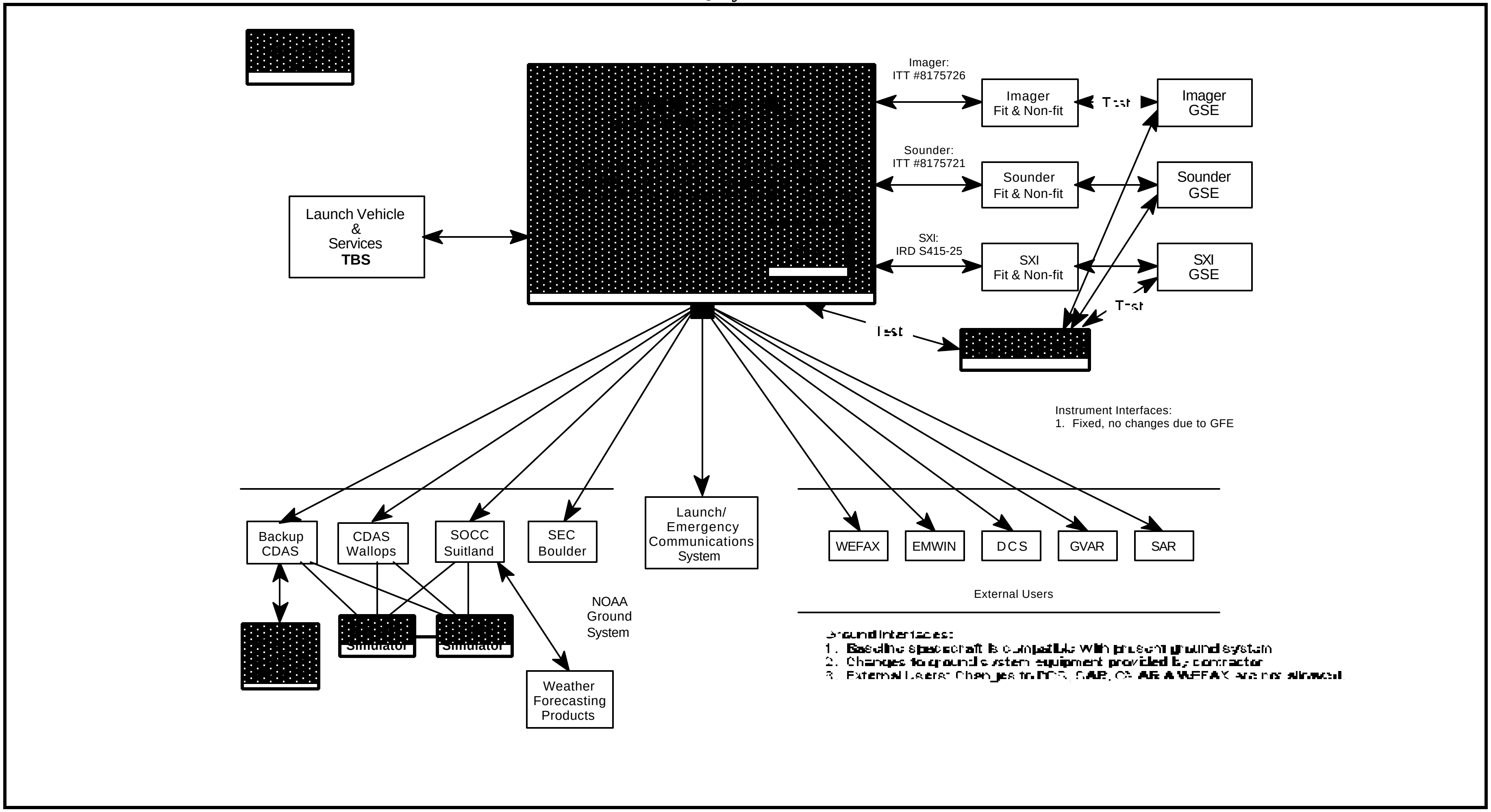
1.1 Mission Phases

The GOES N-Q mission is divided into four phases as follows:

1. Pre-launch (PREL) - The pre-launch phase provides for the design, construction, integration, and testing of each spacecraft as well as the development, installation, and testing of any associated ground equipment software, and facilities required during all subsequent mission phases. This phase begins at contact award and continues until L-1 day.

2. Launch and Orbit Raising (LOR) - The launch and orbit raising phase consists of the launch, transfer orbit operations, appendage deployments, orbital maneuvers to the checkout station, and spacecraft bus functional checkout. The spacecraft contractor is responsible for performing all activities in this

Figure 1.0
GOES N-Q System Interfaces



phase at the government operations center (see section 4.6). This phase begins at L-1 day and continues until the contractor performs an engineering handover to NASA. This phase lasts no more than 25 days. Launch vehicle/services are TBS.

3. Post Launch Test (PLT) - The post-launch test phase consists of all activities required to verify compliance with this document. This includes spacecraft subsystem, instrument, and ground system characterization, instrument calibration, and product assurance. The spacecraft contractor will provide support to assist NASA in performing the PLT. This phase begins at the engineering handover from the spacecraft contractor and continues until the spacecraft is declared operational (accepted) by NASA and handed over to NOAA. This phase lasts approximately five months from the engineering handover review.

4. Operations (OPS) - The operations phase consists of the mission life where the data products are generated and distributed to the user community. This phase begins after PLT and continues until NOAA decides to decommission the spacecraft. This phase includes any periods when the spacecraft is in on-orbit storage.

2.0 APPLICABLE DOCUMENTS

The documents listed in this section are those referenced in the various sections of this specification. The documents establish detailed specifications, requirements, and interface information necessary for the performance of the GOES N,O,P,Q contract. Unless otherwise specified, the document version in effect at the time the contract is executed shall apply. The order in which the documents are listed is not intended to imply any precedence:

- Statement of Work for the GOES-N,O,P,Q, S-415-23, June 1997.
- NASA/GSFC Document S-415-28.
- Fastener Integrity Requirements, GSFC S-313-100.
- Contract Data Requirements List for the GOES-N,O,P,Q Program, S-415-26, June 1997.
- Program Review Requirements for the GOES-N,O,P,Q Program, S-415-27, June 1997.
- Interface Control Document for the GOES-N,O,P,Q Imager; ITT #8175726, 13 Mar. 1997.
- Interface Control Document for the GOES-N,O,P,Q Sounder, ITT #8175751, 1997.
- Interface Control Document for the Solar X-ray Imager, S-415-25, Mar. 1997.
- Interface Control Document for Imager and Sounder Test Equipment and the NO/PQ Spacecraft Ground Support Equipment/Facilities and Spacecraft Launch Facilities (GSE-ICD), ITT #8175788, Draft-Revision B, Aug. 1997.
- GOES NO/PQ CDRL 4-12, Imager Instrument Navigation and Registration Performance Analysis Summary, ITT, Jan. 1997.
- GOES Program - DRL 504-11, Earth Location User's Guide, NOAA/NESDIS, May 1997.
- FED-STD-209.
- JPL Document 810-5, Rev D.
- MIL-STD-461C.
- MIL-STD-462.
- MIL-STD-1246.
- MIL-STD-1522A.
- MIL-STD-1541A.
- MIL-STD-1553B.
- NHB-8071.1.
- MSFC-SPEC-522B.
- CCSDS Recommendations for Radio Frequency and Modulation Systems, CCSDS 401, Sept. 1989.
- ASTM E1417, Standard Practice for Liquid Penetrant Examination.
- International Telecommunications Union (ITU) Radio Regulations, Article 28, 1982 Edition.
- Atlas Launch System Mission Planner's Guide, Lockheed Martin Commercial Launch Services, Inc., Revision 6, Feb. 1997.
- Delta III Payload Planners Guide, McDonnell Douglas Aerospace, Apr. 1996.
- Design Guidelines for Assessing and Controlling Spacecraft Charging Effects, NASA Technical Paper 2361.
- Charged Particle Radiation Exposure of Geostationary Orbits for the GOES N,O,P,Q Satellite Program, GSFC Document X-900-97-004.
- The Astronomical Almanac, U.S. Naval Observatory.
- Meeus, J., Astronomical Formulae for Calculators, Willman-Bell, Inc.
- GOES I-M Telemetry and Command System (GIMTACS) Functional Specification, NOAA/ NESDIS, 22 Feb. 1988).
- GOES I-M Telemetry and Command System (GIMTACS) User's Guide, LMSMS&S.

- GOES IJK/LM GOES Engineering Analysis System (GEAS) Architecture, Operations, and Maintenance Manual, NOAA/NESDIS, May 1997.
- GOES IJK/LM GOES Engineering Analysis System (GEAS) User's Guide, NOAA/NESDIS, Apr. 1997.
- GOES IJK/LM DEChub 900 MultiSwitch and HUBwatch User's Guide, NOAA/NESDIS, May 1997.
- TACTS Overview and Configuration Manual; Version 3.1, Westinghouse Electric Corporation (WEC), Apr. 1991.
- Interface Definition Document for GOES I-M TACTS, Version 2.0, WEC, Mar. 1989.
- *TACTS Operator Manual, Version 2.0, WEC, 20 Feb. 1991.
- GOES IJK/LM Operations Ground Equipment (OGE), DRL 504-02 - Operations Ground Equipment (OGE) Interface Specification - Part 1, NOAA/NESDIS, Apr. 1997.
- GOES IJK/LM Operations Ground Equipment (OGE) Operations and Maintenance Manuals, DRL 504-06, NOAA/NESDIS:

  Part 2 of 22, Sensor Processing System (SPS) Hardware Manual, Feb. 1997.
  Part 17 of 22, Sensor Processing System (SS) Software Maintenance Manual, Feb. 1997.
  Part 11 of 22, Sensor Processing System (SPS) User's Manual, Apr. 1997.
  Part 4 of 22, Product Monitor (PM) Hardware Manual, ISI, Aug. 1994.
  Part 18 of 22, Product Monitor (PM) Software Maintenance Manual, ISI, Nov. 1994.
  Part 12 of 22, Product Monitor (PM) User's Manual, Draft, ISI, Oct. 1996.
  *Part 10 of 22, Dynamic Interaction Diagnostic (DID) System Hardware Manual, Apr. 1997.
  *Part 20 of 22, Dynamic Interaction Diagnostic (DID) Software Maintenance Manual, Apr. 1997.
  *Part 14 of 22, Dynamic Interaction Diagnostic (DID) User's Manual, Apr. 1997.
  Part 7 of 22, Orbit and Attitude Tracking System (OATS) Hardware Manual, Feb. 1997.
  *Part 22 of 22, Orbit and Attitude Tracking System (OATS) Software Maintenance Manual, Rev C, SS/L, Aug. 1995. (New draft expected in July 1997.)
  Part 16 of 22, Orbit and Attitude Tracking System (OATS) User's Manual, Feb. 1997.
- *Rehosted OATS Requirements Overview, NOAA/NESDIS, 30 November 1994.
- Mewe and Groenschild, Astronomical Astrophysics Supplemental Series, Vol 45, pp 11-52, 1981.
- GOES Contamination Analysis Final Report.

(*Note: Refer to Appendix B for specific exempt parts to the indicated documents
CCRS: 6036A. 6037A, 6040A, 6035A, 6039, and 6042)

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3.0 SPACECRAFT GENERAL REQUIREMENTS

3.1 Spacecraft Lifetime

3.1.1 ***Terrestrial Storage*** - The fully integrated and tested spacecraft shall meet the lifetime requirements of section 3.1.2 and 3.1.3 following terrestrial storage, under controlled conditions, for up to five years.

3.1.2 ***On-Orbit Storage*** - The spacecraft design and implementation shall allow for two cumulative years of on-orbit storage. These two years include any time required to prepare for entry into storage and all testing to be performed upon exit from storage. Details of the on-orbit storage requirements are in section 4.2.6.

3.1.3 ***Mission Lifetime*** - The spacecraft, its consumables and life limited items shall be designed and tested to assure the ability to provide five cumulative years of mission lifetime operating within the specifications of this document. These five years shall include the time for PLT, but not the time the spacecraft is in on-orbit storage.

3.2 Operational Concept

The GOES N-Q spacecraft will be launched from Cape Canaveral, FL with the launch vehicle contractor, spacecraft contractor, and government project office participating. The spacecraft contractor's flight operations team shall perform all launch, orbit raising, and bus functional checkout activities with NASA participation. Additionally, the spacecraft contractor shall be responsible for all ground network and communications required to perform the launch and orbit raising activities. All launch and orbit raising activities shall be performed from the selected government operations center (see section 4.6). Once geosynchronous orbit is achieved, all activities shall be performed using the CDAS at Wallops Island, VA as the primary ground station, and a TBD CDAS as the backup.

The GOES operational constellation will consist of two spacecraft, one located at 75° west longitude and the second at 135° west longitude. A third spacecraft may be maintained as an on-orbit spare nominally positioned between 90° and 110° west longitude. During the transition to the GOES N-Q spacecraft series, the operational constellation may contain GOES I-M spacecraft. Detailed orbital location requirements are provided in section 4.4

The spacecraft will be commanded throughout their mission lifetime from the NOAA SOCC with the ground station radio frequency (RF) interface located at the Wallops CDAS (or the TBD backup CDAS). The spacecraft health and safety telemetry streams are received by the CDAS and ground relayed to the SOCC for processing and monitoring. The raw sensor data is received by the CDAS, processed, reformatted, and rebroadcast through the GOES processed data relay (PDR) transponder. The SOCC monitors the GVAR broadcast image quality and performs landmark measurements for use in orbit and attitude determination, and relays the GVAR data to the National Weather Service (NWS). Low-resolution images generated by the NWS Product Generation and Distribution (PG&D) system from the GVAR and other data are ground relayed to the CDAS for broadcast through the GOES weather facsimile (WEFAX) transponder. The data collection platform interrogation (DCPI) transponder is used on a continuous basis to interrogate some data collection platforms and to distribute a timing signal used by numerous entities. The CDAS also receives all data collection platform reports automatically through the data collection platform reports (DCPR) transponder. The platform data is quality checked at the CDAS and then is distributed to the cognizant agencies. Applicable platform data, low-resolution images, and other weather information are fused into the Emergency Management Weather Information Network (EMWIN) S-band broadcast. The multi-use data link (MDL) transmitter will telemeter special spacecraft attitude and instrument data. Expected payloads will be a Solar X-ray Imager (SXI), instrument servo error data, attitude data, and possibly a Lightning Mapper instrument. The NOAA Space Environment Center (SEC) in Boulder, CO will receive the Space Environment Monitor (SEM) instrument data and associated spacecraft telemetry, as well as SXI data via the MDL. All payloads using the MDL will telemeter housekeeping data along with instrument data for processing or data quality monitoring at the SOCC. The GOES Search and Rescue (SAR) transponder will receive emergency beacon transmissions from civil aircraft and large-class marine vessels for relay to ground stations responsible for coordinating search efforts.

3.3 Redundancy

Active redundancy (on-line), passive redundancy (standby spares), or a combination thereof may be employed in order to achieve the specified lifetime. Redundancy may be employed at the subsystem, component or device level (or any combination thereof). Redundancy shall be employed where appropriate in accordance with the fault tolerance requirements as described in section 3.7. Redundancy switching shall occur by ground command unless the system in question affects spacecraft health and safety. Any autonomous switching shall be overrideable in accordance with section 3.8. The command function shall employ active redundancy, and the telemetry functions shall be redundant; communication functions shall also be redundant. With the exception of the magnetometer, which shall be redundant, the SEM instruments are not required to be redundant.

3.4 Spacecraft Charging and Electrostatic Discharge (ESD)

The GOES N-Q spacecraft shall meet the performance requirements of this specification without permanent degradation due to electrostatic discharge (ESD). The GOES N-Q spacecraft design shall control bulk spacecraft, surface, deep dielectric charging, and internal differential voltages to levels that preclude periodic upsets and permanent component degradation or damage. All test harnesses/wiring and test couplers not removed from the GOES N-Q spacecraft at the completion of integration and test shall be terminated to prevent ESD.

To the maximum extent possible, external surfaces of the GOES N-Q spacecraft shall be highly conductive and electrically connected to the spacecraft frame. Partially conductive surfaces (e.g., paints) applied over a conductive substrate shall have a resistivity-thickness product equal to or less than:

$$\underline{rt < 2 \times 10^9 \text{ ohm-cm}^2}.$$

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where “r” is the material resistivity in ohm-cm, and “t” is the material thickness in centimeters. NASA technical paper 2361 titled “Design Guidelines for Assessing and Controlling Spacecraft Charging Effects” should be used as a guide for the prevention of spacecraft charging.

3.5 Spacecraft Magnetic Field

The spacecraft permanent field measured at the magnetometer sensor shall be less than ±100 nanoTesla (nT) in each axis. Section 8.4.10 addresses requirements for stability of the spacecraft field from stray (electrical current generated) fields and/or moving parts. Unavoidable changes in the spacecraft signature at the magnetometer shall be ground corrected as specified in section 9.1.1.

3.6 Spacecraft Grounding

The spacecraft grounding system shall provide a low noise environment for the payload instruments and spacecraft components as detailed in section 10.5.6. Electromagnetic Interference (EMI) noise shall not degrade performance. Of prime concern is normal mode or common mode noise, as defined in the instrument ICDs, that may enter the instruments via the spacecraft interface and degrade performance. Noise from switching transients originating in power converters or digital logic circuits shall be controlled. Steps shall be taken to prevent noise from entering the visible and IR detector bandwidths.

3.7 Fault Tolerance

No credible single point failure in the spacecraft bus, credible single point failure in the instrument to spacecraft interface, or credible single ground operation error shall permanently preclude the spacecraft from providing operational data products or services. No credible single ground operation error in combination with a credible single latent failure in the spacecraft bus or instrument to spacecraft interface shall permanently preclude the spacecraft from providing operational data products or services. Failure modes shall have backups unless categorized as non-credible, or unless their effects do not permanently preclude the spacecraft from providing operational data products or services. Non-credible failure modes shall be supported by sufficient rationale and design margin to preclude their occurrence.

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3.8 Override of Automatic Functions

The spacecraft design shall be such that all onboard automatic functions, or sequences can be disabled/ enabled, executed, or over-ridden by ground command.

3.9 GFE Compatibility

The spacecraft shall be compatible with the GFE instrument interface requirements described in Table 3.9.

3.10 Single-Event Upsets

The GOES N-Q spacecraft shall be designed so that on-orbit single-event upsets (SEUs) of the spacecraft equipment and SEU induced outages of the Imager, Sounder, and SXI, shall be minimized. As a goal, the spacecraft shall be designed so that the SEU induced rate of outages shall be less than one outage in seven years. An outage is anything that prevents execution of instrument operations during normal on-orbit mode or anything that prevents the transmission or relay of instrument science data.

Table 3.9
GFE Instrument Compatibility Requirements

Payload Instrument	Interface Control Document
Imager	ITT document #8175726
Sounder	ITT document #8175751
Solar X-Ray Imager (SXI)	GSFC document #S-415-25
Advanced Imager	TBD
Advanced Sounder	TBD
Lightning Mapper (LM)	TBD

3.11 Radio Frequency (RF) Breakdown and Multipactor

Damage or measurable degradation due to RF breakdown shall be prevented by design. Components with high voltage systems having special outgassing requirements in ground test and on-orbit shall be identified and designed to ensure successful outgassing before activation in the vacuum environment. The design shall preclude measurable degradation due to multipactor and corona in RF systems that must operate during the launch and ascent stages (e.g., filters, switches, and antenna elements), at critical pressures, or in a vacuum environment.

3.12 Workmanship Standards

Flight hardware shall be fabricated in accordance with the requirements of NHB 5300.4 (3A-2, NAS 5300.4 (3H), NAS 5300.4 (3J1), NHB 5300.4 (3L), and NAS 5300.4 (3M). In lieu of using these workmanship standards and procedures, the spacecraft contractor may utilize internal procedures provided they are consistent with the intentions of the NASA standards and produce hardware of the same quality level meeting the GOES mission requirements. Continuing compliance with this requirement shall be demonstrated by the spacecraft contractor during periodic government audits.

3.13 Fasteners

The spacecraft contractor shall comply with the procurement documentation and test requirements for flight hardware and critical ground support equipment fasteners contained in GSFC S-313-100, Fastener Integrity Requirements. In lieu of using these workmanship standards and procedures, the spacecraft contractor may utilize internal procedures provided they are consistent with the intentions of the NASA standards and produce hardware of the same quality level meeting the GOES mission requirements. Fasteners made of plain carbon or low alloy steel shall be protected from corrosion. When plating is specified, it shall be compatible with the space environment. On steels harder than RC 33, plating shall be applied by a process that is not embrittling to the steel.

3.14 Yaw Flip

The spacecraft contractor shall provide the capability for the spacecraft to perform a biannual flip about the yaw axis, where the yaw axis is defined as the nadir-pointing axis, such that the north face of the spacecraft points south. The capability shall include all system modifications required to support the yaw flip and maintain instrument-derived products. Requirements associated with the yaw flip for INR performance, operational performance, the attitude control system (ACS), and the Spacecraft Support Ground System (SSGS) are in the respective sections for these areas. The yaw flip capability shall be provided as an operational feature that may be exercised at the government's option.

3.15 Launch Vehicle Compatibility

The GOES N-Q spacecraft shall be compatible with the Atlas and Delta launch vehicle families as described in Revision 6 of the Lockheed Martin Commercial Launch Services, Inc. *Atlas Launch System Mission Planner's Guide*, and the McDonnell Douglas Aerospace *Delta III Payload Planners Guide*, respectively.

4.0 OPERATIONAL REQUIREMENTS

4.1 Image Navigation and Registration

4.1.1 *General Requirements*

4.1.1.1 ***Overall Performance*** **-** The following INR specification is for the end-to-end system performance, including the spacecraft, Imager, Sounder, and the SSGS. The spacecraft and SSGS performance, when combined with the GFE instruments, shall meet the system INR requirements.

The INR specification values are defined such that 99.7% of the measured, calculated and/or determined values are within the specified range; this is equivalent to the three sigma (ó) value for a Gaussian distribution. All INR specifications apply to the east-west (E-W) and north-south (N-S) performance independently, where E-W is taken as parallel to the Earth's equator and N-S as perpendicular to the Earth's equator at the nominal nadir position.

4.1.1.1.1 ***Selection of Fixed Grid Mode*** **-** During any operational period, either the fixed grid mode or the dynamic gridding mode may be selected independently for the Imager and Sounder. Selection of the fixed grid mode shall result in a fixed grid, where every instantaneous field of view (IFOV) in the selected image/sounding area is assigned to a predetermined pixel/sounding location. Each pixel/sounding within a 65^{o} Earth central angle (ECA) shall be Earth located and registered (frame-frame and within-frame) to the specifications of sections 4.1.2, 4.1.3, and 4.1.4. INR requirements for pixels/soundings between 65^{o} and 70^{o} ECA shall not be degraded to more than twice the requirements of sections 4.1.2, 4.1.3, and 4.1.4. Selection of the fixed grid mode shall not affect any other INR requirements or any instrument radiometric requirements. In the dynamic gridding mode, all INR requirements shall be met except for within-frame registration (section 4.1.3) and frame-frame registration (section 4.1.4).

4.1.1.1.2 ***Resampling*** - Resampling of Imager and Sounder radiometric data to meet any INR performance requirement is not permitted. However, pixel shifting to align IR channels with the Imager visible channel or the Sounder Star channel with the IR channels is permitted to achieve INR performance requirements.

4.1.1.2 ***INR Data***

4.1.1.2.1 ***INR Data Availability*** **-** Earth located and registered data from the Imager and Sounder meeting all the requirements of this specification shall be continuously provided, independent of any other instrument, except as noted in sections 4.1 and 4.2. Each pixel/sounding and its associated Earth location data shall be available for retransmission through the GOES spacecraft within three minutes after that pixel/sounding is sensed.

4.1.1.2.2 ***INR Data Transmission*** **-** Earth location data for the entire image/sounding area shall be part of the GVAR processed data stream. All INR requirements shall be met without requiring changes to the GVAR data. It shall be unnecessary to process data from the telemetry or MDL data streams to produce calibrated, gridded, and/or formatted data for distribution to users.

4.1.1.3 ***Instrument Operations*** - The time required for star sense operations in support of the Imager INR function shall be limited to less than 135 contiguous seconds per 30-minute interval. The time required for star sense operations in support of the Sounder INR function shall be limited to less than two 120-second periods per 60-minute interval. Other than star sensing, stationkeeping and housekeeping operations as detailed in section 4.2, spacecraft operations shall not interrupt or in any way preclude continuous Imager and Sounder scan operations.

4.1.1.4 ***Solar Eclipses*** - The INR shall operate but need not meet the INR requirements in this specification during the period starting 0.5 hours before and ending 1.5 hours after the penumbra period for solar eclipses of the spacecraft lasting 72 minutes or less.

4.1.1.5 ***Lunar Events*** - INR performance shall not be degraded by any lunar events, except for lunar eclipses of the Sun lasting more than 72 minutes, during which INR may be disabled. On the day following an eclipse with a duration exceeding 72 minutes, the INR shall operate but need not meet the INR requirements in this specification for a time duration starting 23.5 hours after the eclipse started and ending 25.5 hours after the eclipse ended.

4.1.1.6 ***Coregistration*** - Instrument coregistration performance shall not be degraded by the spacecraft.

4.1.1.7 ***Event Time Determinations*** - The current capability of the GOES I-M SSGS to determine the start and end times of all events potentially affecting INR operation and/or instrument performance shall be preserved and augmented if required.

4.1.1.8 ***Recovery Following Yaw Flips*** - The time required to recover INR-specified performance following a yaw flip (i.e., north face of spacecraft rotated to be the south face) shall not exceed 24 hours.

4.1.2 ***Navigation (Earth Location) Requirements*** - Navigation requirements are defined with respect to the instrument scan angle. The corresponding Earth located navigation requirement is for a pixel/-sounding sample at nadir. For pixel/sounding samples not at nadir, the Earth location error shall not exceed F(â) times the allowable error at nadir, where â is the Earth central angle measured between the subsatellite point and the centroid of the pixel/sounding sample. Values of F(â) are given in Table 4.1.2.

4.1.2.1 ***Imager Navigation*** - During a normal operational period a pixel shall be Earth located to within 56 µradians (2 km at nadir).

4.1.2.2 ***Sounder Navigation*** - A sounding sample shall be Earth located to within 280 µradians (10 km at nadir).

4.1.3 ***Within-Frame Registration (WIFR)*** - The WIFR requirements apply only to the fixed grid mode of operation.

Table 4.1.2
Earth Location Error Factor

Earth Central Angle (Degrees)	**Expansion Factor F(â)**
5	1.01
10	1.02
15	1.06
20	1.10
25	1.17
30	1.25
35	1.37
40	1.52
45	1.71
50	1.99
55	2.37
60	2.95
65	3.89
70	5.69
75	10.41
80	65.53

4.1.3.1 ***Imager WIFR*** - The variation from the expected nominal angular separation between the centroids of any two pixels in the same image in the E-W or N-S scan directions shall be as given below, where the nominal angular separation is:

[measured adjacent pixel angular separation] • [pixel separation count].

4.1.3.1.1 ***Visible and IR Channels*** - During a normal operational period, the WIFR shall be within 42 μradians.

4.1.3.1.2 ***Imager E-W Line-Line Shear*** - The combined effects of the spacecraft and ground system shall not add more than 20 μradians to the E-W separation between the centroids of corresponding pixels in two adjacent visible scan lines.

4.1.3.1.3 ***Imager N-S Line-Line Shear*** - The combined effects of the spacecraft and ground system shall not add more than 20 μradians to the N-S separation between the centroids of corresponding pixels in two adjacent visible scan lines.

4.1.3.2 ***Sounder WIFR*** - The following requirements shall be met for any sounding taken during a 120-minute operational period. In the determination of the Sounder registration performance, the portion of the error caused by systematic, predictable, and reproducible effects of pixel rotation as the Sounder points north or south of the nadir position shall not be included as a part of the WIFR error.

During a normal operational period, the variation from the expected nominal angular separation between the centroids of any two sounding samples from the same viewing area in the E-W or N-S scan directions shall be within 84 μradians.

The nominal angular separation for the Sounder is the:

[measured adjacent sounding angular separation] • [sounding separation count].

4.1.4 ***Frame-Frame Registration (FFR)*** - The FFR requirements apply only to the fixed grid mode of operation.

4.1.4.1 ***Imager FFR*** - The E-W or N-S angular registration of corresponding pixels in repeated images of the same selected Earth area shall be as given below.

4.1.4.1.1 ***Imager FFR - 15 Minute Separation Period*** - The FFR within a 15-minute period during a normal operational period shall be within 28 μradians.

4.1.4.1.2 ***Imager FFR - 90 Minute Separation Period*** - The FFR within a 90-minute period during a normal operational period shall be within 42 μradians.

4.1.4.1.3 ***Imager FFR - 24 Hour Separation Period*** - The FFR for corresponding pixels in images of the same selected Earth area taken during normal operational periods shall be within 112 μradians.

4.1.4.2 ***Sounder FFR*** - The E-W or N-S angular registration of corresponding pixels in repeated soundings of the same selected Earth area shall be as given below.

4.1.4.2.1 ***Sounder FFR - 90 Minute Separation Period*** - The Sounder FFR within a 90-minute period during a normal operational period shall be within 84 μradians.

4.1.4.2.2 ***Sounder FFR - 24 Hour Normal Operational Periods*** - Within any 24-hour period, the Sounder FFR for corresponding soundings of the same selected Earth are taken during normal operational periods shall be within 224 μradians.

4.1.5 ***INR Telemetry Requirements*** - INR engineering telemetry not required for routine operations shall be provided to permit on-orbit testing, calibration, and troubleshooting. This telemetry shall be transmitted on the MDL attitude data channel. Examples of this telemetry include signals from the ACS, INR compensation signals, and signals to permit on-board tuning of INR specific functions.

4.1.6 ***INR Post-launch Diagnostic Capabilities***

4.1.6.1 ***INR Checking*** - The capability to monitor all INR related compensation and ACS signals in real-time shall be provided. The corresponding sampling rate shall be at least four times the bandwidth of the monitored signal (e.g., a signal with a 1 Hz bandwidth shall be sampled at a minimum rate of 4 Hz, or every 0.25 seconds).

A ground monitoring capability to confirm that the spacecraft is generating and applying the correct compensation signal(s) and analysis capabilities to combine the various signals to isolate/identify anomalies shall be provided.

4.1.6.2 ***Performance Determination*** - The following capabilities shall be provided to facilitate INR anomaly resolution:

4.1.6.2.1 ***Angular Displacement Sensor (ADS)/Angular Velocity Sensor (AVS)*** - The capability to measure angular displacement from 2.3 Hz to 200 Hz in three axes to an accuracy of 0.2 µradians 1ó per axis for angular displacement amplitudes < 20 Fradians and to an accuracy of 2.0 Fradians 1ó per axis for angular displacement amplitude <200 Fradians, at the spacecraft-Imager and the spacecraft-SXI interfaces shall be provided. A second range of 1.0 to 1000 µradians or 0 to 50 mrad/sec shall be selectable by command. This data shall be sampled at least once every 1.25 milliseconds and sent to the NOAA SOCC via the MDL. The frequency response of each ADS/AVS unit shall be measured during unit level testing in the frequency range of 0.1 Hz to 200 Hz.

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4.1.6.2.2 ***INR Pointing Errors*** - The capability to quantitatively determine the contributions to the INR in-orbit performance from the spacecraft, Imager, Sounder, and SSGS shall be provided. The INR performance requirements to be determined are: navigation, within-frame registration, and frame-frame registration. As a minimum, it shall be possible to determine the respective contributions to the INR performance errors due to the following:

1. Spacecraft control system noise.
2. Spacecraft control system errors.
3. Spacecraft dynamic interactions.
4. Instrument-induced attitude disturbances (e.g., caused by blackbody calibrations or SXI motions).
5. Errors in the estimation of the various required compensations (e.g., curve fit errors for rapidly changing thermal induced pointing errors).
6. Errors resulting from differences between the desired compensation of the instrument line-of-sight and the resulting instrument applied compensation.
7. Instrument pointing errors associated with the servo performance.
8. Thermally induced non-repeatable errors (spacecraft & instruments).
9. Orbit determination errors.
10. Attitude determination errors.
11. Estimates of non-repeatable errors (magnitude & characteristics).

This determination capability may be provided through additional sensors, hardware and/or software in the spacecraft and SSGS. The 3ó accuracy of the determination shall be demonstrated to be within ¼ of a pixel for measurements related to landmarks, and within 4 µradians E-W and 7 µradians N-S for stars and all other measurements; statistical averaging may be used.

4.1.6.3 ***INR Calibration*** - The current calibration capabilities to determine spacecraft motion caused by instrument blackbody calibrations, perform star tracking for indefinite periods, and image the moon shall be continued and maintained. Additional calibration capabilities shall be provided, if required, to initialize, calibrate, and/or maintain the INR performance.

4.1.7 ***INR Budget Summary*** - For the purpose of developing the INR system level budgets, Table 4.1.7-1 contains the INR budget allocations for the Imager; and Table 4.1.7-2 contains the INR budget allocations for the Sounder. The GFE instrument (BOL) Measured/Analytic values in the tables shall be used in the determination of the system level errors. More detailed information is provided in ITT CDRL 4-12 for the GOES-NO/PQ Imager Instrument.

In addition to the random pointing errors (shown in the tables) introduced by the instruments in the determination of star and landmark locations, the associated 3ó ground determination of a star location (i.e., error with no other noise) shall be: #8 µradians E-W, and #14 µradians N-S. For Imager visible and IR landmarks, the E-W and N-S 3ó ground errors shall be # 0.5 pixel (i.e., 8 and 14 µradians for visible, and 32 and 56 µradians for IR pixels). Sounder landmarks are not currently used in the INR process.

Table 4.1.7-1
INR Budget Allocations - Imager (3ó in µradians)

Requirement	**System Specification**	**GFE Instrument (BOL) Measured/Analytical E-W/N-S**	**GFE Instrument (BOL) Specification E-W/N-S**
Navigation	56	19.4/12.1	±25/20
Within Frame Registration (Reg.)	42	27.5/17.2	±33/28
Frame-Frame Reg. 15 minute 90 minute 24 hour 48 hour	 28 42 112 N/A	 20/12.1 20/12.1 20/12.1 -	 ±28/28 ±28/28 ±28/28 ±28/28
Visible to IR Coregistration (CEP)	N/A	±50*	±50

* Diurnally compensated in the GOES I-M SSGS to be less than ±28 µradians.

The day-to-day variation in the instrument thermal profiles can be assumed to be no worse than 10 µradians. A worst case set of instrument profiles are included in Figures 4.1.7-1 and 4.1.7-2, which are available in electronic form. These profiles are derived from ITT provided data, and modeling of the Imager deformations with the sun at 23° north latitude for the N-S deformation and 23° south latitude for the sun E-W deformation. The N-S curve results from combining the roll and roll misalignment, and the E-W curve from combining the pitch and pitch misalignment mispointings.

Table 4.1.7-2
Sounder 3ó INR Budget Allocations (in µradians)

Requirement	**System Specification**	**GFE Instrument (BOL) Measured/Analytical E-W/N-S**	**GFE Instrument (BOL) Specification E-W/N-S**
Navigation	280	24.1/11.5	±30/30
Within-Frame Registration	84	43.4/16.2	±42/42
Frame-Frame Registration Short-term 24 hour	 84 224	 19.3/10.8 -	 - ±42/42
Star Channel-to-Channel 8 Coregistration (CEP)	N/A	-	±36

Figure 4.1.7.1
E-W Thermal Pointing Variation - Instrument Worst Case

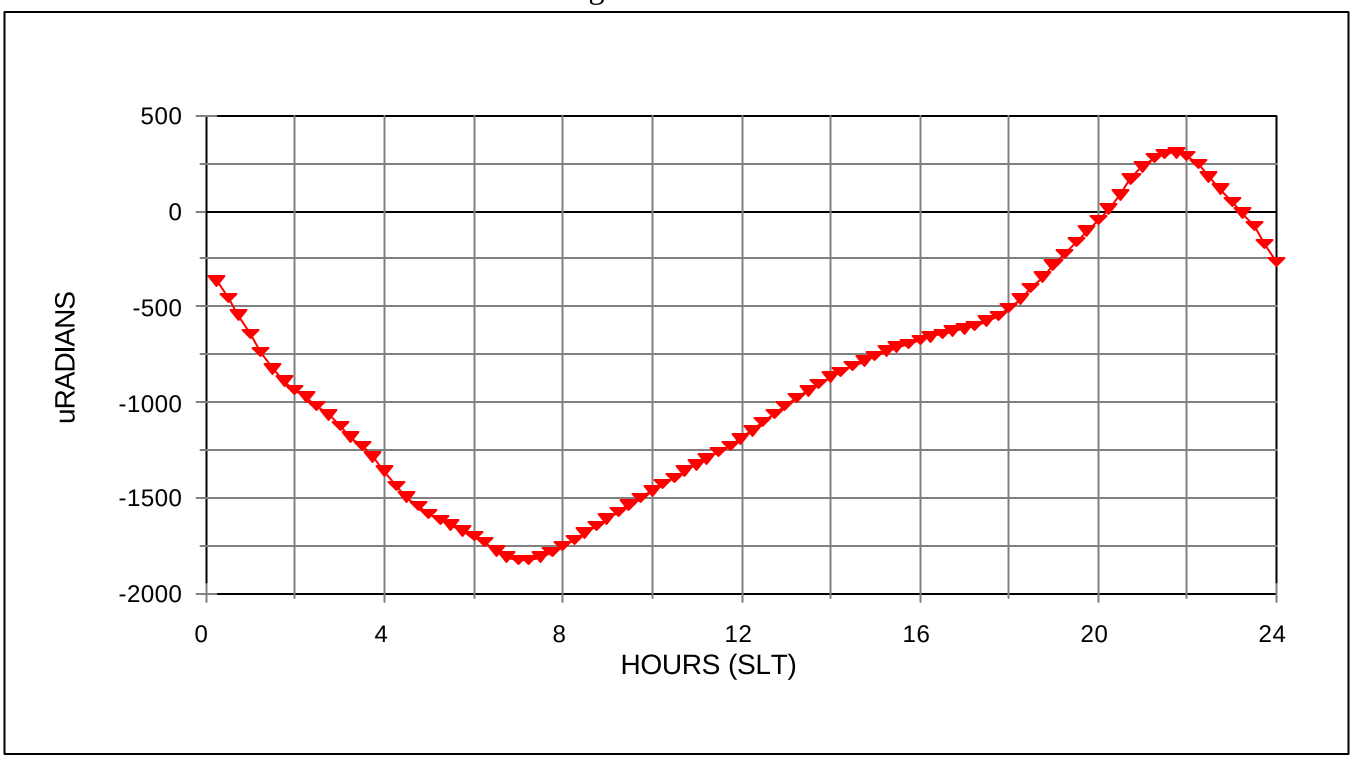
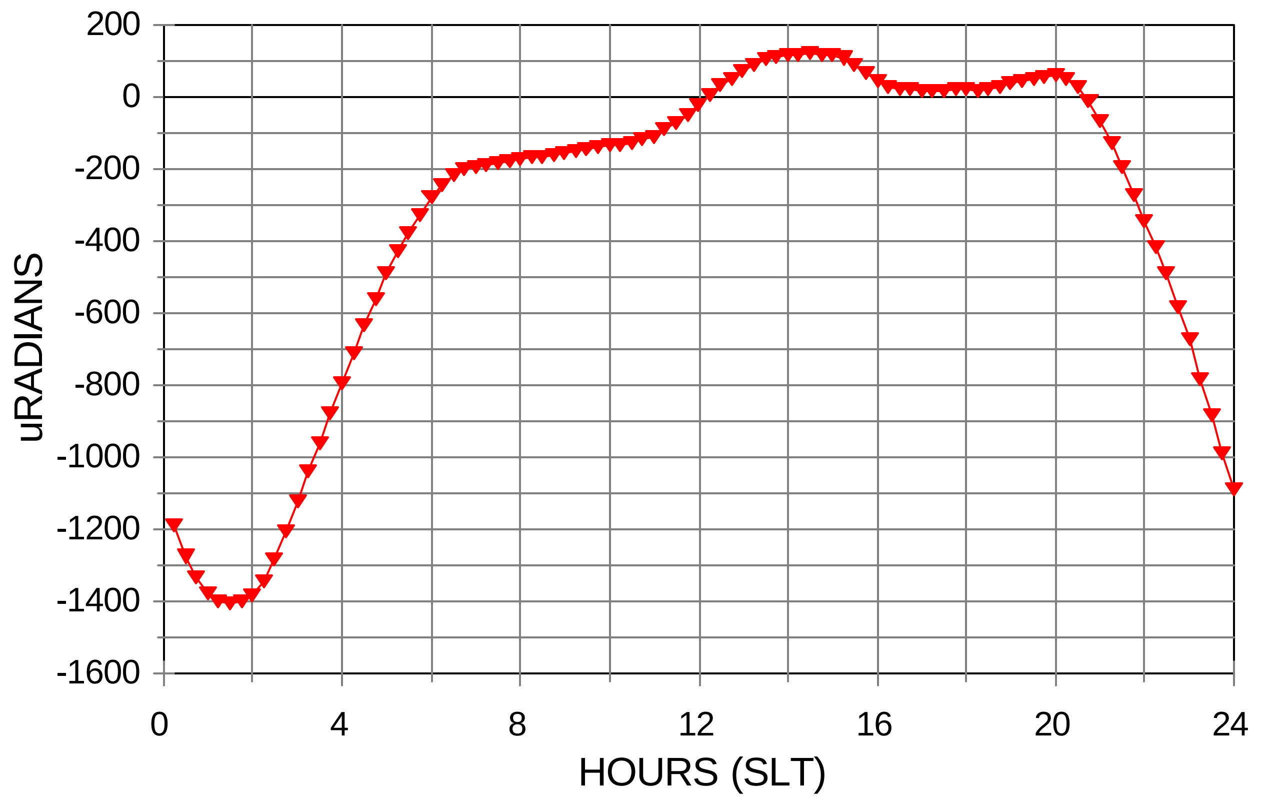


Figure 4.1.7.2
N-S Thermal Pointing Variation - Instrument Worst Case



4.2 Modes of Operation

The GOES N-Q spacecraft shall have pre-defined, commandable modes of operation. Each combination of spacecraft sensors and actuators for spacecraft control shall be utilized in a separate spacecraft mode (sub-mode). Since normal operations place a high demand on spacecraft resources from the user community, interruption of normal spacecraft operations will be allowed only to the extent given in Table 4.2. While the spacecraft contractor shall determine the exact operational modes required to perform the mission as per the GOES N-Q SOW (S-415-23) section 2.3.2, the following subsections describe the various modes needed.

4.2.1 ***Launch Mode*** - While in this mode, the instrument optical axes shall not point at the sun within the constraints specified in section 3.2.3.6a of the Imager ICD and section 3.2.3.7a of the Sounder ICD.

4.2.2 ***Normal Operations Mode*** - This mode is used at all times when the instruments are performing the mission. Telemetry and command and instrument uplinks and downlinks shall occur simultaneously as per sections 10.1 and 10.2. Equinox shall not significantly change the spacecraft configuration, and shall not interrupt any spacecraft and instrument operations. Solstice shall not change the spacecraft configuration and shall not interrupt spacecraft or instrument operations. Imager and Sounder outgassing may occur in this mode. Outgassing shall not affect other spacecraft subsystem operations.

4.2.3 ***Stationkeeping Mode*** - This mode shall be used for any large-scale orbital maneuvers requiring a long period (hours) before INR returns to specification. All maneuvers shall produce the least possible impact to normal instrument operations.

4.2.3.1 ***Stationkeeping Maneuver Period*** - The maximum duration of a stationkeeping maneuver period shall be # 1.5 hours. Imager and Sounder data need not be collected during the maneuver period.

4.2.3.2 ***Stationkeeping Recovery Period*** - The maximum duration of a stationkeeping recovery period during which INR need not meet all the requirements of this specification shall be # 6 hours. Imaging data shall be collected, but the data need not be Earth located or registered to the requirements of this specification.

4.2.4 ***Housekeeping Mode*** - This mode shall be used to perform daily engineering housekeeping operations on the satellite bus. These operations shall be limited to a maximum of two per day with a maximum duration of 10 minutes each. Disruptions to instrument operations and/or impacts to INR performance shall be limited to the 10-minute housekeeping period.

4.2.5 ***Safe Hold Mode*** - This mode shall be a non-operational mode for the instruments. It shall preserve the health and safety of the spacecraft and instruments in the event of an anomalous condition, as described in section 10.3.4. Commanding requirements shall be as defined in section 10.1. Entry into safe hold shall not cause damage, degradation, or reduction of mission life to any spacecraft subsystem or instrument. Entry into safe hold mode shall be initiated via ground command or by the on-board computer.

4.2.6 ***On-Orbit Storage Mode*** - The on-orbit storage mode shall be a non-operational mode that minimizes the operation of life-limited components, subsystems, and fuel. The on-orbit storage mode shall not require ground interaction more frequently than every 30 days, excluding anomalous events and spacecraft telemetry monitoring. The intent of this mode is to minimize commanding, monitoring, and maneuvers. The GOES N-Q spacecraft shall be capable of being placed into on-orbit storage within 24 hours. The GOES N-Q spacecraft shall be ready for post-storage testing within 24 hours from call-up by NOAA. Post-storage testing shall last no more than 45 days. Spacecraft and instrument engineering telemetry shall be downlinked while in this mode, and there shall be no telemetry nulls in excess of those
described in section 10.1. There shall be no spacecraft operation during the on-orbit storage mode that causes the instrument optical axes to point at the sun in violation of the constraints specified in sections 3.2.3.6a and 3.2.3.7a of the Imager and Sounder ICDs, respectively. Utilization of this mode shall in no way reduce or diminish the mission life requirements of section 3.1.3.

4.2.7 ***End-of-Life Mode*** - The End-of-Life mode shall allow for boosting the spacecraft to a supersynchronous orbit 350 km above geostationary orbit. The spacecraft shall not radiate power (to avoid interference with operational spacecraft at the same longitude), and shall not generate debris.

Table 4.2
Operational Time Allotment Summary

Mode	Item	Requirement
Normal Operations	Duration (per hr. excluding SK, HK)	
	Imaging[1]	55.5 min.
	Imager Star Sense	4.5 min.
	Sounding[1]	56 min.
	Sounder Star Sense	4 min.
	SXI Imaging[1]	10 sec./min.
	SXI Idle Time	50 sec./min.
Stationkeeping	Duration (includes Setup)	# 1.5 Hrs
	Recovery Time[2]	# 6 Hrs
	Number of Allowable Occurrences	
	East/West	# 6 per year
	North/South	# 1 per year
Housekeeping	Duration	10 min.
	Recovery Time[2]	0
	Number of Allowable Occurrences	2 per day
Yaw Flip	Yaw Flip Maneuver & Attitude Recovery	# 1 hr
	Full INR Compliance[2]	# 24 hrs

[1] Operating within INR specifications.
[2] Time until INR is back within specification.

4.3 Simultaneous Operations

Each spacecraft functional requirement shall perform within specification regardless of the mode of operation of any other functional requirement. No spacecraft function shall interfere with the performance of any other function, except for the disruptions identified in section 4.2.

4.4 Geosynchronous Orbit and Locations

4.4.1 ***Operational Orbital Locations*** - The GOES N-Q spacecraft shall meet all specifications in this document at any geostationary altitude, with a target inclination of 0°. The expected target locations will be 75° west longitude for GOES East and 135° west longitude for GOES West. Once a given orbital station is chosen and attained, the GOES N-Q spacecraft shall meet the specifications in this document for any station within ±0.5° longitude of that desired station at an orbital inclination of 0.0 ±0.5°.

4.4.2 ***Orbital Location (Station) Changes*** **-** GOES N-Q spacecraft shall be capable of nine station location changes during the GOES N-Q on-orbit life defined in section 3.1. The station changes will be:

1). From checkout location to an on-orbit storage location.
2). From the on-orbit storage location to the operational station location.
3,4,5). Three changes of operational station location while remaining within INR specifications at #1°/day drift.
6,7). Two emergency relocations at #3°/day drift.
8). From operational station to end-of-life longitude.
9). Boost from geosynchronous altitude at end-of-life longitude to end-of-life supersynchronous altitude in accordance with section 4.2.6.

The spacecraft shall be capable of maintaining a drift rate during station changes of up to 3° longitude per day without functional performance degradation, and up to 1° per day without degradation to any instrument and/or INR performance specification.

4.4.3 ***Maximum Required Operability Ranges*** **-** GOES N-Q shall be capable of basic operations at locations between 5° west and 145° west longitude, and inclinations of up to 8.0°. This is intended as an end-of-life or emergency option and nothing on the spacecraft shall preclude operations in these extreme locations. It is not expected that INR will be met under the wider constraints of this section.

4.5 Autonomous Equipment Reconfiguration

All autonomous equipment reconfigurations initiated by the onboard processor shall be flagged in real-time normal telemetry stream. All autonomous functions shall incorporate a ground-commandable override capability which will prevent the function from executing when override is enabled. Non-overridable equipment reconfigurations shall only be used to avoid spacecraft or component health and safety hazards. Autonomous reconfigurations shall not be required for normal operations of the spacecraft or its components. The spacecraft shall perform all reconfigurations required to enter into the safe hold mode.

4.6 Government Operations Center

The spacecraft contractor shall provide for a transparent switchover from the government operations center (GOC) to the SOCC at any time following handover at geosynchronous orbit, but before the end of PLT.

5.0 ENVIRONMENTAL REQUIREMENTS (ON-ORBIT)

The spacecraft shall meet the requirements of this specification after exposure to the environments described below:

5.1 Radiation Environment

Adequate radiation protection shall be provided by means of local shielding and/or spacecraft structure to assure reliable performance of all components in the spacecraft for the lifetime of the spacecraft. The spacecraft radiation environment is defined by GSFC document X-900-97-004, titled Charged Particle Radiation Exposure of Geostationary Orbits for the GOES N,O,P,Q Satellite Program.

5.1.1 ***Total Dose Radiation Level* -** The total dose radiation level each electronic part is expected to encounter over the lifetime of the GOES N-Q spacecraft shall be calculated taking into account the shielding effects of the spacecraft materials and geometry. As a minimum, all parts shall be capable of meeting the specification requirements after exposure to two times the calculated total dose radiation level.

6.0 LAUNCH VEHICLE, SERVICES, & SUPPORT REQUIREMENTS

6.1 Spacecraft Contractor Provided Launch Vehicle and Services

The spacecraft contractor shall provide a Delta or Atlas class launch vehicle for each GOES N-Q spacecraft. NASA shall approve the launch vehicle selected for each spacecraft using the following acceptance criteria:

1. GOES N-Q spacecraft shall not be the first flight on a new or significantly modified existing launch vehicle.
2. The launch vehicle shall have a demonstrated flight record of reliability (looking back up to 5 years) which has been verified to meet predicted vehicle and performance parameters (e.g., within 3ó criteria).
3. For integrated spacecraft and launch vehicle operations (including launch phase), requirements defined in the spacecraft-specific section of this specification (S-415-22) shall flow down to the launch vehicle.
4. The launch services shall be United States provided, and the launch site shall be located at Cape Canaveral, Florida.
5. No co-manifested or secondary spacecraft are permitted unless authorized by the Contracting Officer.

6.2 Launch and Orbit Raising Phases

The spacecraft shall provide telemetry and be configured to receive commands during the entire LOR period. The spacecraft shall be capable of executing critical operations via stored absolute time commands and real-time commands sent from the ground.

6.2.1 ***Launch and Orbit Raising*** - The spacecraft configuration shall be verifiable at the government operations center during the entire launch countdown. All stored command loads shall be verifiable during the countdown. Data availability in the LOR period shall be subject only to ground station line-of-sight constraints and the coverage requirements of paragraph 10.1.4.

6.2.2 ***Special Subsystem Requirements for LOR*** - The spacecraft contractor shall provide for the following during:

1. Telemetry and command - transmission starting from L-4 hours.
2. Propulsion - commanding required to perform subsystem calibration.
3. Flight Software - dump complete image of on-board software.
4. Energetic Particle Sensor - power on as soon as possible after launch and calibrate.
5. Magnetometer - turn on the instrument before deploying the magnetometer boom. See section 9.1.1 for requirements pertaining to on-orbit zero offset determination and calibration. Magnetometer turn-on and deployment shall occur after orbit raising activities are completed and before handover. CCR6044 B Mod 36
6. Imager - commanding for contamination avoidance and heater configuration, as required.
7. Sounder - power on the filter wheel at launch, and perform commanding for contamination avoidance and heater configuration, as required.

6.3 Network Compatibility and Testing

The spacecraft shall be compatible with both the launch and on-orbit operations networks.

6.4 Mission Operations Simulations Support

While the spacecraft is in the launch base payload processing facility and at the launch pad, the spacecraft and test equipment shall be capable of providing real-time telemetry data to either the government operations center or the SOCC, whichever is being used to support the launch, for mission operation simulations.

7.0 SPACECRAFT SUPPORT GROUND SYSTEM

7.1 General Requirements

The spacecraft contractor shall provide a ground system to support the requirements of the as-built GOES N-Q spacecraft. The GOES N-Q SSGS shall provide, as a minimum, the functionality of the GOES I-M SSGS (i.e., OATS, MDL Processing System (MPS), Dynamic Interaction Diagnostic (DID), GIMTACS, TACTS, PM, SPS), but is not constrained to use the same architecture. All GOES N-Q SSGS elements shall use a common time reference to permit accurate time correlation of data between elements. Considering the potential to minimize floor/desktop space requirements and minimize operator training, the spacecraft contractor shall, if feasible, use, adapt, modify, or upgrade GOES I-M SSGS elements for use in the GOES N-Q SSGS, as stipulated in the remainder of this section, and provide new elements where reuse is not feasible. Where GOES I-M SSGS elements are reused for GOES N-Q, backward compatibility of GOES N-Q software is not required; rather, what is required is that workstations and processors be configurable by menu selection to load and start up either GOES N-Q or any other software at boot-up. Note: GOES I-M SSGS software will be made available as GFE; and no GOES I-M SSGS hardware will be provided as GFE, but access to this hardware will be made available by the government for test and integration purposes; and 3) the capabilities of the GOES I-M SSGS must not be degraded by the introduction of the GOES N-Q SSGS hardware or software into the SOCC and CDAS environments.

7.1.1 ***GOES N-Q Telemetry and Command System (GTACS)* -** The spacecraft contractor shall provide a telemetry and command processing system (GTACS) to support the as-built GOES N-Q spacecraft. The GTACS shall provide, as a minimum, the system functional capabilities and an equivalent level of operability and performance as specified in the GOES I-M Telemetry and Command System (GIMTACS) Functional Specification (dated 22 February, 1988) and the GOES Engineering Analysis System (GEAS) Architecture, Operations, and Maintenance Manual.

Currently, VAX 3100 workstations are capable of being booted up to run either GIMTACS or Polar Acquisition and Command System (PACS) software, allowing the same workstations to support launch and engineering activities for both programs. These workstations will also be used to support the Integrated PACS (IPACS) system when the US Air Force Defense Meteorological Satellite Program (DMSP) operations are transferred to NOAA. Design work is ongoing for replacement GIMTACS/ PACS workstations, with a hardware solution expected in the fourth quarter of 1997. Because of the multiple system usage and to avoid hardware duplication, these replacement workstations shall be used for GOES N-Q support. Adequate resource margin requirements will be imposed on the replacement workstations to support expected GTACS requirements and alternative operating system environments (e.g., VMS and UNIX). Workstation replacement documentation will be distributed as it becomes available.

7.1.2 ***GOES N-Q Telemetry Acquisition and Command Transmission System (NTACTS)* -** The spacecraft contractor shall provide new Telemetry Acquisition and Command Transmission Systems (TACTSs) capable of supporting the as-built GOES N-Q spacecraft. These new GOES N-Q TACTS (NTACTS) shall retain the system functional capabilities and equivalent level of operability and performance specified in the TACTS Overview and Configuration Manual; Version 3.1 (dated April 1991), the TACTS Interface Definition Document, Version 2.0 (dated March 1989), and the TACTS Operator Manual, Version 2.0 (dated February 20, 1991). An exception to the above requirements is that the NTACTS does not require the current capabilities of displaying spacecraft telemetry and generating commands to the spacecraft, as currently provided in the GOES I-M TACTS.

7.1.3 ***Sensor Processing System (SPS)*** - The SPS consists of Image Processing Systems (IPSs) that process the raw instrument data and analyst workstations used for calibration and engineering analysis. There are four IPSs at the Wallops CDAS with one planned for the backup CDAS. There are three analyst workstations at the SOCC, two at the Wallops CDAS, and one is planned for the backup CDAS. The existing SPSs shall be used to process the raw GOES N-Q Imager and Sounder data, generate the GVAR formatted data stream, and perform instrument calibration and analysis. Any spacecraft contractor changes to the SPS functionality or interfaces, in particular the changes required by the yaw flip, shall be the responsibility of the spacecraft contractor and will require approval by NASA. The GVAR format shall not be modified, unless it can be guaranteed that users who do not choose to modify their GVAR ingest systems will still be able to receive all the data in the current GVAR. The government will provide the spacecraft contractor access to an SPS for test, integration, and validation purposes. The OGE Interface Specification, DRL 504-02 and the SPS User's and Software Maintenance Manuals describe the SPS architecture, capabilities, and interfaces.

7.1.4 ***Product Monitor (PM)*** - The spacecraft contractor shall utilize the as-built GOES I-M PMs for the monitoring and analysis of GOES N-Q GVAR data and to obtain landmark data. Currently, there are four PMs at SOCC, three at the Wallops CDAS, and one is planned for the backup CDAS. Replacement PMs are currently under procurement, with initial delivery expected in the first quarter of 1999. For the purposes of this request for proposal, the existing PM interface specifications given in the OGE Interface Specification, DRL 504-02 should be used. Any spacecraft contractor changes to the PM functionality or interfaces, as specified in the OGE Interface Specification and the PM User's and Software Maintenance Manuals, shall be the responsibility of the spacecraft contractor and shall require NASA approval. The government will provide the spacecraft contractor access to a PM for test, integration and validation purposes.

7.1.5 ***Orbit and Attitude Tracking System (OATS)*** - The spacecraft contractor shall provide an orbit and attitude determination (OAD) system for the as built GOES N-Q spacecraft with, as a minimum, the functionality and operability of the GOES I-M OATS, as described in the OGE Interface Specification, and the OATS User's and Software Maintenance Manuals. If feasible, the existing GOES I-M OATS Digital Equipment Corporation (DEC) Alpha workstations shall be used for the GOES N-Q OATS.

7.1.6 ***MDL Receive System, MPS, and MPS Server*** - The GOES I-M MPS is an extension of the existing Sun workstation-based DID and SPS analyst workstation capability currently under development by NOAA contractors to support the GOES-M SXI in addition to the GOES-8 and GOES-10 MDL. As defined in the GOES I-M documentation, the MPS includes servers, DID and SXI telemetry handling , and the DID and SXI analysis capabilities. Because of the expected redesign of the MDL receive system and changes to the MDL data streams, the GOES N-Q MPS comprises only the data handling/analysis process that will run on the existing Sun analyst workstations. The DID, MPS server, and receive system functions are identified as separate components. See the SPS and DID Hardware, Software and User's Manuals, and the OGE Interface Specification for a description of the DID and SPS analyst workstation capabilities and interfaces, and the DID archive design.

The spacecraft contractor shall provide a new GOES N-Q MDL receive system and MPS server to demodulate, bit synchronize, provide instrument and system status data exchange with GTACS, and archive the GOES N-Q spacecraft MDL and SXI data streams described in section 10.2, and the SXI ICD. The spacecraft contractor shall have the option to either design the MPS server function to interface with the GOES I-M MPS analyst workstation capability currently under development, or to develop a GOES N-Q MPS analysis function compatible with the GOES N-Q MPS server capable of running on the existing analysis workstations. The spacecraft contractor shall also provide a GOES N-Q DID function to process the GOES N-Q MDL attitude data. The GOES N-Q DID shall be integrated into the existing SOCC SPS/MPS analyst workstations.

7.1.7 ***GOES SSGS Network*** **-** The GOES I-M SSGS Network (and the NOAA Polar Environmental Satellite (POES) ground network) consists of 10BaseT Ethernet local area networks (LANs) with T1-derived wide area network (WAN) circuits (diversely routed) interconnecting the SOCC and CDASs. DEChub 900 Multiswitch hubs at the SOCC, the Wallops CDAS, and the planned backup CDAS provide the capability to define virtual LANs and the flexibility to move devices among the virtual LANs. The WANs employ Lucent Technologies T1 multiplexers and DEC RouteAbout routers. The SOCC and Wallops CDAS each have separate GIMTACS and PACS DEChubs interconnected via a transponder module to permit extending LANs across both DEChubs.

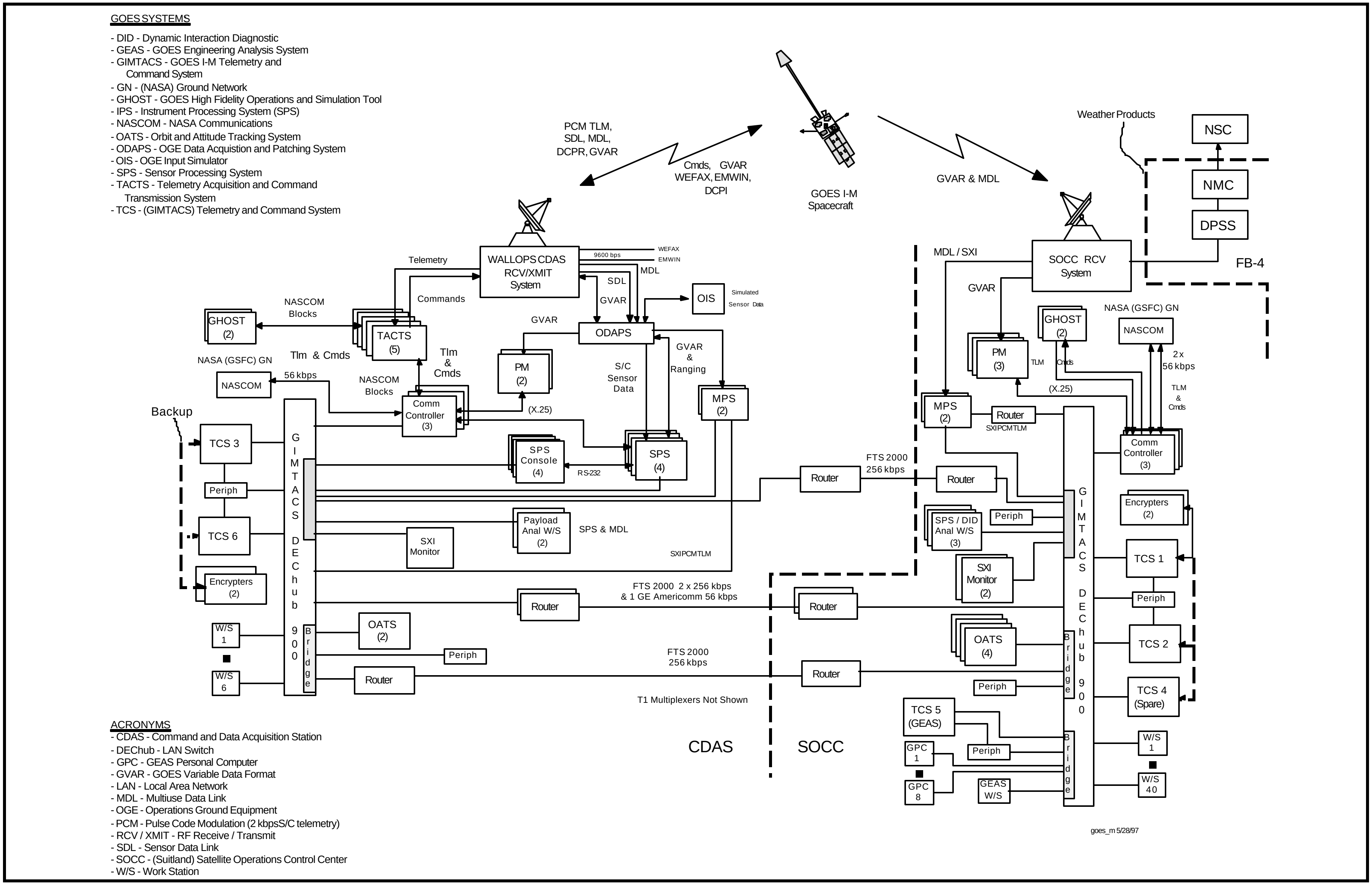
The spacecraft contractor shall augment the existing SSGS Network DEChub equipment to support GOES N-Q inter- and intra- system communications and shall provide connectivity to the GIMTACS DEChub to allow devices to be moved between GOES N-Q LANs and the GOES I-M, PACS, and IPACS LANs. The DEChub 900 MultiSwitch and HUBwatch User's Guide, and the DEC Knows Networks product literature describe the currently available DEChub components. The OATS Hardware Manual also describes the SSGS Network and the OATS interfaces to it. The SOW (S-415-25) provides a list of DEChub equipment required for a 10 Mbps Ethernet implementation as a guide since SSGS Network is not well documented. Figure 7.1.7 illustrates the GOES I-M SSGS as it is expected to look in the GOES-M time frame, minus the backup CDAS. In particular, the figure illustrates the various LAN segments and the WAN connectivity between the SOCC and the Wallops CDAS. The shaded boxes indicate LAN segments independent of or bridged to the GIMTACS LAN.

7.2 Performance Requirements

7.2.1 ***GOES N-Q Telemetry and Command System (GTACS)*** **-** The GTACS shall reside at the SOCC in Suitland, the WCDAS, and the backup CDAS. The GTACS shall:

1. Be compatible with all telemetry and command interfaces with the as-built GOES N-Q spacecraft, GFE instruments, and the NTACTS.
2. Include encryption/decryption devices compatible with the new GOES N-Q command decryption equipment.
3. Be compatible with modified and unmodified GOES I-M SSGS elements used in support of the GOES N-Q spacecraft and instruments, and comply with the interfaces specified in the Operations Ground Equipment (OGE) Interface Specification, DRL 504-02.
4. Be sufficiently flexible to accommodate new and changed pulse code modulated (PCM) telemetry processing and commanding requirements imposed by advanced Imager and Sounder instruments (see sections 9.2 and 9.3), and a Lightning Mapper (LM) instrument.
5. Process two PCM health and safety telemetry data streams simultaneously, a DSN stream and a CDA stream, at the real-time reception rate of both, plus the Imager and Sounder instrument wideband telemetry data from the SPS, and the SXI and other instrument wideband telemetry data plus any additional spacecraft health and safety telemetry from the MDL Processing System (MPS) server.
6. Process NTACTS receiver AGC telemetry data at an update rate equal to that of the fastest PCM telemetry point.

Figure 7.1.7
GOES I-M SSGS in the GOES-M Time Frame



7. Process telemetry and distribute to operator workstations within 0.5 seconds of its receipt at the control center, distributing telemetry updates at the same rate as telemetry blocks/minor frames are received from the spacecraft.
8. Process and output clear mode and encrypted commands at the maximum allowable rate of the as-built N-Q spacecraft command receivers, outputting commands issued by a commanding workstation within one telemetry block/minor frame update period.
9. Support the scheduling, commanding, and telemetry processing requirements of the yaw flip maneuver.
10. Fail over to redundant systems within one minute; in the case of failure of an operator position commanding a spacecraft, switchover of any non-command mode operator position to command mode within 15 seconds of operator initiation of the action.
11. Support up to 100 user workstations, each capable of performing real-time commanding and telemetry monitoring, spacecraft operations scheduling, and off-line telemetry analysis functions.
12. Provide sufficient capacity to support telemetry and command processing for eight real and simulated GOES N-Q spacecraft concurrently.
13. Provide processing system resource margins of 50% (e.g., CPU speed, RAM and disk capacities) for every component (excluding the GIMTACS operator workstations) when supporting three fully operational spacecraft configurations and using the full capabilities of all components (the spacecraft contractor can assume the GIMTACS/PACS replacement workstations will have this capacity). GTACS components shall have this resource margin at final delivery before the GOES-N launch.
14. Provide redundancy such that no single point of failure, workstation outage, or system performance degradation in the GTACS will disrupt or preclude real-time telemetry and command processing operations.
15. Use the GOES I-M Archive System for the on-line storage of the GOES N-Q DSN and CDA (2209 MHz and 1694 MHz) PCM telemetry streams, NTACTS AGC data, Imager and Sounder wideband telemetry received from the SPS, and SXI data received from the MRS&S. The interface between GTACS and the GOES I-M archive system shall be in accordance with the GOES N-Q GTACS to GOES I-M Archive System ICD TBD. CCR 4183C MOD 36
16. Provide an SSGS configuration monitoring function to monitor the status and send configuration commands to all GOES N-Q SSGS component systems, similar to the GIMTACS configuration page.
17. Provide an operator workstation function meeting the following general requirements (and design goals, given that reuse of the replacement GIMTACS workstations is required):
    a. Windows-based graphical user interface (GUI).
    b. Startup time from completion of cold boot in three minutes or less.
    c. Minimum of 20 active windows to include plots, telemetry display pages, etc., while maintaining a CPU resource margin goal of 50% at final delivery before the GOES-N launch.
    d. Resizable windows minimizable to an icon, with selectable font sizes to permit zooming and full page width views of open windows.
    e. Concurrent off-line analysis or scheduling function operation without disruption to real-time telemetry monitoring functions.
    f. Hard copy page snap capability for any active window.
    g. On-line, context sensitive help and an on-line copy of the user's manual.
    h. User input/response logging.
    i. Text messaging capability to address any connected GTACS, SPS, PM, or GOES N-Q OATS workstation. For those systems, such as the PM, without this capability, the spacecraft contractor shall define the message structure.
    j. Text file output to local/system printer.
    k. Workstation hard disk space warning alert program.
    l. Visual and audible alarm upon loss of a telemetry stream being monitored.
18. Provide the following specific real-time operator/engineer (Ops/Eng) workstation capabilities:

a. Real!time spacecraft commanding restricted to one workstation per spacecraft at any one time.
b. Command authority takeover by any workstation signed on to that same spacecraft within 30 seconds, with notification sent to all other workstations monitoring the spacecraft.
c. Monitoring the telemetry and command activities of a minimum of three spacecraft concurrently while in command mode.
d. Up to 10 simultaneous real- and non-real-time plots, each displaying up to four mnemonics, with the following characteristics:
    1. Non-real-time (archive data) plots to screen limited to one-hour of data
    2. Continuously scrolling real-time plots (i.e., not time limited)
    3. Plot widths definable in hours or number of points (typically one hour for mnemonics updating every 0.512 seconds, or 7000 points for subcommutated mnemonics)
    4. Plot a user-defined telemetry point reference curve (e.g., mnemonic average value and standard deviation during a user-specifiable period derived from archive data) as an overlay to a single-mnemonic real-time plot
    5. Print to screen and hard copy devices.
e. Access to off-line telemetry archive to define and submit batch archive plots.
f. Hyperlinks to the operational database for mnemonics on plots and page displays to open a pop-up window displaying the mnemonic's database record (descriptor, yellow and red limits, EUs, calibration curve, etc.).
g. Telemetry display page and real-time plot definition utility.
h. Local copy of the telemetry and command database, command procedures, limit sets, contingency operations procedures (COPs), and other global data.
i. Hyperlinks to spacecraft subsystem block diagrams and Contingency Operations Procedures (COPs) associated with critical alarms.
j. Automatic sensing of database version changes and notification to the operator.
k. Audible and visual alarms for telemetry out-of-limit conditions.

19. Provide the following off-line analysis and trending capabilities:
    a. Generation of 24-hour plots of all values for one telemetry parameter within one minute, and for six telemetry parameters within two minutes, with hard copy output available from a printer within one minute afterward. <u>These timing requirements take effect upon receipt by the trending/analysis process of the requested archive data.</u> CCR4183C Mod 36

b. Batch (scheduled) and interactive (non-scheduled) plot definition and submission from any GTACS operator position.
c. Prioritization of plot requests to favor interactive plot requests.
d. Monochrome and color hard copy capability.
e. User-defined mathematical operations on multiple user-specified telemetry parameters (e.g., an equation processor).
f. Generation of user-defined plots with up to six y-axis data sets versus a single x-axis data set on a single grid. It shall be possible to plot time on either axis. When time is plotted on the x-axis, it shall be specified by the user for any time frame from ½ the period of the fastest telemetered data to 30 days. The x-axis units shall be user-selectable and, at minimum, in units of seconds, minutes, hours, days or weeks.
g. It shall be possible to put up to eight of these grids on an 8-½" by 11" page. Each grid on the page shall include the date and time the data was submitted for initial processing and the actually plotted. The page shall also identify the data set or file from which the data was extracted. User-defined optional data fields consisting of at least primary and secondary titles 80 characters long.
h. If data thinning methods are employed, the minimum, maximum, and time integrated average of the data points shall be saved for the time span over which the data was thinned. The thinning method shall be selectable to be over a time period or over a number of points.
i. Data shall be tagged with questionable or static data flags, and the user shall have the option to include or exclude questionable and static data. An optional notation shall be provided to indicate the fraction of good data in the selected data set.
j. Data gaps shall be indicated on plots by penups..
k. Generation of tabular telemetry value (TVAL) reports in file and hard copy form.
l. Support other capabilities detailed in GEAS documentation.
m. Black and white, and color laser print capability equivalent to the existing SOCC QMS-1725 and QMS-MCX-1 printers, respectively.

20. As a minimum, provide the GOES I-M Scheduler Workstation capabilities specified in the GIMTACS Functional Specification and the following specific capabilities:
    a. Import, create, edit, store, and print spacecraft command procedures (CPs).
    b. Create time-tagged 24-hour command schedules for spacecraft stored-command uploads and commanding from GTACS Ops/Eng positions. These schedules shall be generated using raw commands and CPs, and shall take into account special instrument operations, such as calibrations, star looks, etc., and solar/lunar exclusion zones.
    c. Verify the accuracy of schedules to ensure no commands inimical to the health and safety of the spacecraft and instruments are included.
    d. Provide schedule templates to permit building special schedules (e.g., rapid and super-rapid scans) rapidly for ground and stored program use, such that only particular parameters as scan coordinates and execution time need be provided for activation.
    e. Print CPs, schedules, and reports at a workstation or line printer.
    f. Provide a file management utility to browse, select, delete, edit, rename, copy, and protect CPs, schedules, and reports.
    g. Provide listings of the contents of CPs, schedules, and reports via printers or workstation screen display.
    h. Perform instrument scan coordinate conversions.
    j. <u>Provide schedule shadowing on the ground that will give visibility into the execution of onboard scheduling commands (including Instrument Objects), support parallel telemetry monitoring on the ground, and allow operator to stop on-board schedule execution and resume schedule execution on the ground.</u> CCR 6008 Mod 24

21. The ground system shall provide the capability to calucate the length of telemetry outages and generate an audible alarm when outages exceed a user defined limit.

7.2.2 ***GOES N-Q Telemetry and Command Transmission System (NTACTS)*** - The NTACTS shall provide the interface between the radio frequency (RF) receive systems at the WCDAS, and the GOES N-Q Telemetry and Command Processing System (GTACS). In addition to the general requirements stated in sections 7.1 and 7.1.2, the NTACTS shall:

1. Be capable of simultaneously supporting both the CDA and the DSN PCM telemetry streams, and the command uplink of one GOES N-Q spacecraft.
2. Support the telemetry and command interfaces with the GTACS via a direct Ethernet connection using the Internet Protocol (IP) format being adopted by the NASCOM IP transition project.
3. Provide receiver automatic gain control (AGC) data to the GTACS via a direct Ethernet connection using IP, as in item 2 above, and with an update rate equal to that of the fastest PCM telemetry point.
4. Reconfigure from a standby mode to active support of a GOES N-Q spacecraft within one minute.
5. Have no single point of failure or system performance degradation capable of disrupting or precluding real-time telemetry and command processing operations.
6. Provide the capability at WCDA and BUCDA to generate and receive DSN ranging tones through the DSN transponder. CCR 4073 Mod 24

7.2.3 ***GOES N-Q OATS*** - In addition to the general requirements stated in sections 7.1 and 7.1.5, the GOES N-Q OATS shall:

1. Retain, as a minimum, the system functional capabilities and equivalent level of operability and performance as specified in the Rehosted OATS Requirements Overview (dated 30 November, 1994), and the Operations Ground Equipment (OGE) Operations and Maintenance Manuals, DRL 504-06, Part 16 of 22 - User Manual, Orbit and Attitude Tracking System.
2. Comply with the SPS and PM interfaces specified in the Operations Ground Equipment (OGE) Interface Specification, DRL 504-02.
3. Have a sufficiently flexible and modular design to accommodate any new data handling, and image navigation and registration (INR) requirements imposed by advanced Imager and Sounder instruments and a LM instrument.
4. Support the INR requirements imposed by the yaw flip maneuver.
5. Use a distributed system architecture with the processing and storage capacity to fully support the simultaneous processing of five spacecraft configurations from a single workstation.
6. Provide a minimum system processing resource margins of 50% when supporting full operations for two spacecraft. This resource margin shall be effective at final delivery before the GOES-N launch.
7. Perform the orbit and attitude determination computation for one spacecraft within four minutes.
8. Provide hot-standby computer systems.
9. Automatically detect and alarm within two minutes the failure of any prime workstation, and shall switch over to a backup workstation within one minute.
10. Provide the capability to monitor graphically INR performance in real-time for each spacecraft.
11. Provide the capability to generate statistical reports, tables, histograms, and trending plots outlining short-term and long-term INR performance for each spacecraft.
12. Provide a COTS “point and click” software capability to simplify data manipulation and graphical data analysis.
13. Provide a black and white, and color laser print capability equivalent to that of the SOCC QM-MCX-1 laser printe

14. Perform the functions necessary to meet system level INR performance and diagnostic requirements defined in section 4.1.
15. Provide the capability to compute spacecraft orbit using CDA ranging data. CCR 4073 Mod 24

7.2.4 ***MDL Receive System and MPS Server*** - In addition to the general requirements presented in section 7.1 and subsection 7.1.6, the GOES N-Q MDL receive system and MPS server shall:

1. Provide intermediate frequency (IF) receive systems and computer systems to demodulate, bit synchronize, demultiplex, decommutate, and archive simultaneously the MDL data streams from three GOES N-Q spacecraft.
2. Provide a new or modified DID function to be resident on the MPS workstations to process the GOES N-Q spacecraft MDL attitude data. The GOES N-Q DID shall provide, as a minimum, the system functional capabilities and an equivalent level of operability and performance as specified in the GOES IJK/LM Operations Ground Equipment (OGE) Operations and Maintenance Manuals, DRL 504-06, Part 14 of 22, Dynamic Interaction Diagnostic (DID) User's Manual, Apr. 1997.
3. Comply with the interfaces specified in the GOES SXI to Ground System ICD, the Dynamic Interaction Diagnostic (DID) Hardware and Software Maintenance Manuals, and the Operations Ground Equipment (OGE) Interface Specification.
4. Provide a sufficiently flexible and modular DID design to accommodate, with minimal code changes, any new data handling, data storage, and INR requirements imposed by advanced Imager and Sounder instruments and a LM instrument.
5. Interface with the GTACS for the exchange of SXI PCM telemetry data, any MDL attitude data needed by GTACS or OATS (as determined by the spacecraft contractor), and PCM telemetry data from any additional GOES N-Q instrument or system using the MDL.
6. Provide a circular on-line (i.e., disk) data storage capability for one week of MDL data (including SXI and LM data) for three spacecraft, plus an additional 10 Gigabytes of on-line storage for user work space and save areas.
7. Provide a capability to generate off-line archive tapes automatically or with minimal operator involvement and the capability for users to download selected data sets to tape for export.
8. Provide an archive directory facility capable of accessing all MDL data stored in the on-line archive.
9. Provide a capability to import/recover archive data from off-line tapes.

7.2.5 Digital Wideband Tape Recorders (DWTR) **–** The DWTR shall meet the following requirements:

1. The DWTR shall be able to simultaneously record GOES I-M satellite data including all three wideband data streams: Imager, Sounder, and MDL. The DWTR must be able to replay this data to reproduce the same data streams.

2. The DWTR shall be able to record or replay GOES NOPQ satellite data including four simultaneous data streams: Imager, Sounder, and the two MDL/SXI (CCSDS) interleaved channels. CCR41535C Mod 28

3. The DWTR shall have an upgrade option for recording/replaying a minimum 5 megabites per second combined data rate for the four input and output channels.

4. The DWTR shall be capable of accepting the NRZ-S signal from the electronics boxes of the Imager, Sounder, and SXI directly, for instrument level testing.

5. The DWTR shall record and replay all the bits received, regardless of whether proper frame-sync patterns are included (this is required to allow the unit to capture and reproduce data which includes bit errors).

6. The DWTR shall have a maximum bit-error rate of 1 in 10^{12} bits.

7. The DWTR units with 4 channel upgrade shall be capable of accepting the NRZ-S output of the two demodulators (both I and Q channels). This shall include the Imager and Sounder data streams and both MDL/SXI streams. CCR4135C Mod 28

8. Replayed data from the DWTR shall be in a PCM NRZ-S format identical to the input signal properties. Each stream shall look identical to the stream presented by live instruments or spacecraft to the associated bit sync. These playback streams shall appear identical in content and structure to the original data when presented to each bit sync.

9. The DWTR shall time-stamp the wideband data as it is recorded to a one second resolution. The recorder must have a capability to synchronize its time clock to an external standard, such as GPS time.

10. During replay the DWTR shall make available the timestamps of the data to a one second resolution.

11. The DWTR shall have removable commercial off-the-shelf DLT media and drive units.

12. The DWTR shall maintain a directory of the contents of each recorded volume of wideband data, as part of the information on the media. Utilities shall be available to read and present this directory in playback mode.

13. The DWTR shall uniquely identify each recorded tape (or other media) in such a manner that it can be distinguished from all other tapes recorded on each recorder. This shall allow tapes from different recorders to be intermixed without ambiguity. This unique identification shall be a part of the information on the tape.

14. The DWTR shall include a Java-based user interface, which can be used to fully control all record and playback operations from computers networked to the recorder.

15. The DWTR units with DLT changers shall be able to record data continuously (24 hours per day, seven days per week), subject only to the availability of replacement tape media on a periodic basis.

16. The DWTR shall have options to support either single tape units or multiple tape changer units. When using a tape changer unit, the recorder shall be capable of recording a minimum of 30 hours of continuous data on all channels without operator attention (except for equipment failures) with a single changer, and 60 hours with two tape changers.

17. The DWTR units with DLT changers shall automatically invoke tape-head cleaning as required without operator intervention.

18. The DWTR units with DLT changers shall recover from errors on a single tape without operator intervention or loss of data. If multiple tape errors require operator intervention, then an automated method for notifying an operator shall be provided. CCR4135C Mod 28

19. The DWTR units with DLT changers shall be capable of operating correctly when all but one of the tape units or changers has failed. Changeover from a failed unit to a good unit shall be automatic and not require operator intervention.

8.0 VERIFICATION TEST PROGRAM

8.1 Electrical Function Test Requirements

The following sections describe the required performance verification program functional and performance tests required to verify nominal spacecraft operation before, during, and after environmental testing.

8.1.1 ***Electrical Interface Tests*** **-** Electrical interface testing shall be performed prior to the integration of any assembly, component, or subsystem into the next higher hardware assembly, and to verify that all interface signals are within the acceptable limits of applicable performance specifications.

Prior to mating with other hardware, the electrical harnessing shall be tested to verify proper characteristics, such as routing of electrical signals, impedance, isolation, and overall workmanship.

8.1.2 ***Comprehensive Performance Tests (CPTs)*** **-** A CPT shall be conducted on each hardware element after component, subsystem and spacecraft assembly. When environmental testing is performed at a given level of assembly, additional CPTs shall be conducted at the hot and cold extremes of the temperature or thermal-vacuum test, and at the conclusion of the environmental test sequence. This testing shall include tests at the allowable maximum and minimum bus voltages.

8.1.3 ***Limited Performance Tests (LPTs)*** **-** LPTs shall be performed before, during, and after environmental tests, as appropriate, to demonstrate functional capability has not been degraded by the tests. LPTs may also be used where CPTs are not warranted or practicable. LPTs shall demonstrate the performance of selected hardware and software functions is within pre-defined acceptable limits. The spacecraft contractor shall identify in the verification plan the hardware elements requiring LP testing.

8.1.4 ***Performance Operating Time and Failure-free Performance Testing*** **-** The following operating times shall be used for designing the integration and test program:

1. One-thousand hours of operating/power-on time shall be accumulated on all flight electronic hardware, including both sides of redundant equipment and spares prior to launch.
2. At the conclusion of the performance verification program, the spacecraft shall have demonstrated failure-free performance testing for at least the last 350 hours of operation. The demonstration may be conducted at the subsystem level of assembly when spacecraft integration is accomplished at the launch site and when the 350-hour demonstration cannot practicably be accomplished on the integrated spacecraft. Failure-free operation during the thermal-vacuum test exposure is included as part of the 350 hour demonstration with 100 hours of the trouble-free operation.

The general intent of the above requirements is to accumulate sufficient operating time on all flight hardware. However, it is understood that under certain conditions this requirement may not be met. For example, component failure during spacecraft level thermal vacuum or other tests and hardware change-out just prior to launch may not provide sufficient time to demonstrate these requirements. These interruptions to the operating requirements shall be evaluated on a case-by-case basis taking into account the criticality of the hardware element and the risk impact on achieving mission goals.

8.2 Magnetic Testing

The spacecraft shall undergo magnetic testing consisting of a permanent magnetic moment test and a stray magnetic field measurement test. The permanent magnetic moment test shall provide an unbiased estimate of the dipole moment of the unpowered spacecraft in the three coordinate axes which define the spacecraft geometry. The estimate shall be demonstrated to be repeatable (3 trials) to within the larger of 10% or 1 ampere-meter^2 per axis, peak-to-peak.

The stray magnetic field measurement shall demonstrate at the system level that the magnetic field measurement accuracy required (see section 9.1.1) will not be compromised by field changes associated with any spacecraft operation. Spacecraft systems shall be operated during the test to evaluate magnetic signatures associated with normal on-orbit operations, and to establish the magnitude of magnetic signatures at the flight sensor position.

8.3 End-to-End Testing

A series of end-to-end (ETE) compatibility tests shall be performed during the pre-launch period as defined in SOW section 3.6.2.3. The components utilized for these tests shall be the spacecraft, the SOCC and CDAS ground systems, and the GSFC/NOAA network elements. The ETE tests shall be designed to demonstrate full operational capabilities for each phase of the GOES N-Q mission. The ETE tests shall be conducted from the GOC by the mission operations team.

8.4 Electromagnetic Compatibility Requirements

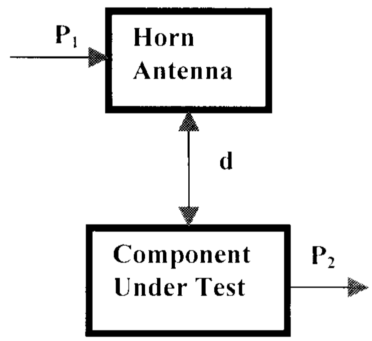
8.4.1 ***General Requirements*** **-** The general requirements for electromagnetic compatibility are as follows:

1 The spacecraft and its elements shall not generate electromagnetic interference that could adversely affect its own subsystems and components, contractor and GFE instruments, or the safety and operation of the launch vehicle and launch site.

2. The spacecraft and its components shall not be susceptible to emissions that could adversely affect their safety and performance. This applies whether the emissions are self-generated or emanate from other sources, or whether they are intentional or unintentional.

3. The EMC test program is meant to uncover workmanship defects and unit-to-unit variations in electromagnetic characteristics, as well as design flaws. The spacecraft-level qualification and spacecraft-level flight acceptance EMC programs are the same. The component-level qualification EMC test program shall comply with the requirements of section 8.4. The component-level flight acceptance EMC test program shall uncover workmanship defects. The component-level acceptance EMC test program for the IPC components shall consist of CE and RE and for the RF components shall consist of RF susceptibility and RF leakage tests as defined below. CCR 6009 Mod24

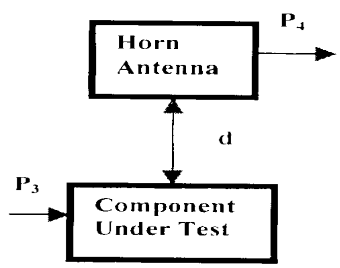
A. **RF Susceptibility Acceptance Testing.** The RF Susceptibility test shall be performed by spraying the RF Component under test with an RF field from a waveguide to coax adapter used as a horn antenna, as shown in the figure 8.4.1-1. The signal at the output of the component under test is compared to the input power to the waveguide adapter. The waveguide adapter is maintained at distance (d) of 0.3 meter from the component under test. For passive RF components this ratio (P_2/P_1) shall be less than –75dB. For active RF components, a reference carrier at the operating point of the component is used as a reference. For active RF components, the ratio (P_2/P_1) shall be consistent with meeting the requirements of section 10.1 and 10.2. CCR 6009 Mod24



CCR 6009 Mod24

Figure 8.4.1-1 RF Component Susceptibility Test

B. **RF Leakage Acceptance Test.** The RF Leakage test shall be performed by sniffing the RF Component under test with an RF field from a waveguide to coax adapter used as a horn antenna, as shown in the figure 8.4.1-2. The signal at the output of the waveguide adapter is compared to the input power to the component under test. The waveguide adapter is maintained at a distance (d) of 0.3 meter from the component under test. For passive RF components this ratio (P_4/P_3) shall be less than –75db. For active RF components, a reference carrier at the operating point of the component is used as reference input. For active RF components, the ratio (P_4/P_3) shall be consistent with meeting the requirements of section 10.1 and 10.2. CCR 6009 Mod24



CCR 6009 Mod24

Figure 8.4.1-2 RF Component Leakage Test

4. The spacecraft and its components shall be compatible with the conducted and radiated EMI performance of GFE instruments as described in their respective ICD or requirements document.
5. The requirements of this section apply to all continuous emissions. Transient or pulse type noise shall be considered on a case-by-case basis, but shall not in any way detrimentally affect the normal operation of the communications or telemetry and command subsystems, or the instruments.

The EMC test requirements herein, when performed as a set, are intended to provide an adequate measure of hardware quality and workmanship. The tests are performed to fixed levels intended to envelope those to be expected during a typical mission and also allow for some hardware degradation during the mission.

Most of the EMC test requirements are based on the requirements of MIL-STD-461C and 462, as amended by Notice 1. All references in this document to MIL-STD-462 assume reference to Notice 1.

8.4.2 ***Launch Vehicle Compatibility*** - Additional EMC requirements may be placed on the spacecraft by the launch vehicle or launch site, so test levels shall be tailored to include the launch vehicle and launch site environment. These EMC requirements shall be established during coordination between the contractor and the launch vehicle contractor.

8.4.3 ***Spurious Signals*** - Spurious signals above specified limits shall be eliminated. Spurious signals below specified limits shall be analyzed to determine if a subsequent change in frequency or amplitude is possible. If possible, the spurious signals shall be eliminated to protect the spacecraft and instruments from the possibility of interference. Retest shall be performed to verify that intended solutions are effective.

8.4.4 ***Testing at Lower Levels of Assembly*** - Testing shall be performed at the component, subsystem and spacecraft assembly levels. Testing at lower levels of assembly has many advantages: it uncovers problems early in the program when they are less costly to correct and less disruptive to the program schedule; it uncovers problems that cannot be detected or traced at higher levels of assembly; it characterizes box-to-box EMI performance, providing a baseline that can be used to flag potential problems at higher levels of assembly; and it aids in troubleshooting.

8.4.5 ***Conducted Emission Requirements***

8.4.5.1 ***Narrowband Conducted Emissions*** - The narrowband conducted emissions on the instrument (42V) power and power-return leads shall be limited to the levels specified in Figure 8.4.5.1. Testing shall be in accordance with MIL-STD-461C and 462, test numbers CE01 and CE03, with limits as shown in Figure 8.4.5.1.

8.4.5.2 ***Broadband Conducted Emissions*** - Broadband conducted emissions on the instrument (42V) power, and power-return leads shall be limited to the levels specified in Figure 8.4.5.2. Testing shall be in accordance with MIL-STD-461C and 462, test number CE03, with limits as shown in Figure 8.4.5.2.

8.4.5.3 ***Common Mode Noise Conducted Emissions*** - A conducted emissions test to control common mode noise (CMN) shall be required for the spacecraft. This frequency domain current test shall be performed on all non-passive components which receive or generate 42V instrument spacecraft primary power. The CMN limit requirements are described in Figure 8.4.5.3. The CMN test procedure is the same as the narrowband CE01/03 tests, except that the current probe is placed around both the plus and return primary wires together.

Figure 8.4.5.1
Narrowband Conducted Emission Limits on Spacecraft Power Lines

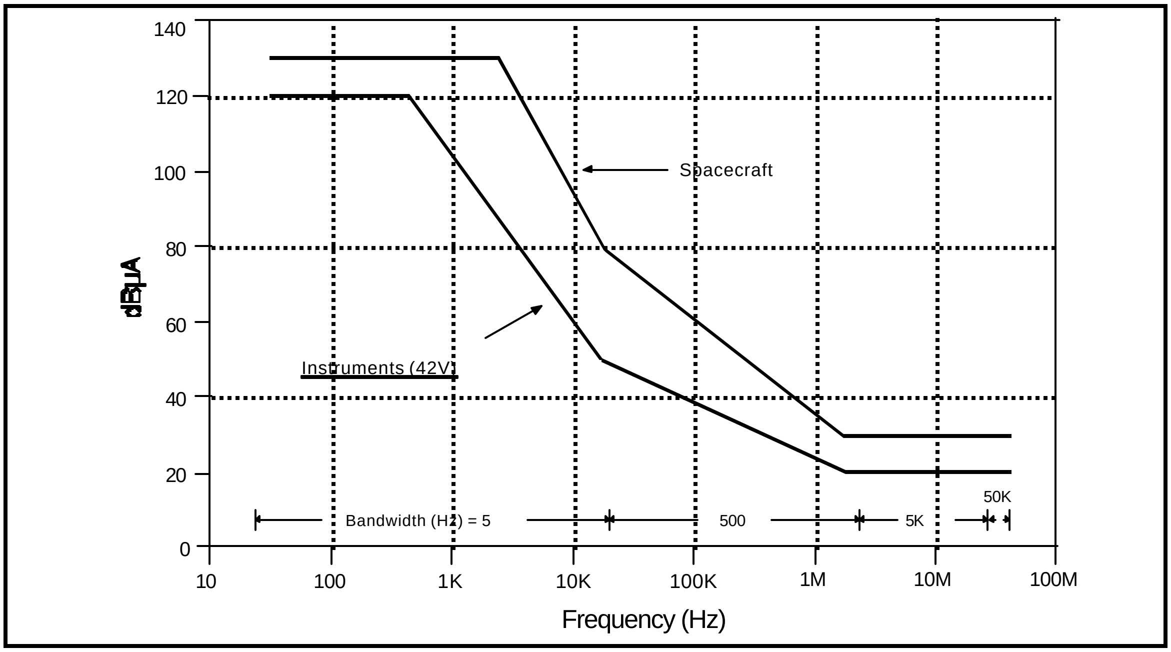


Figure 8.4.5.2
Broadband Conducted Emission Limits on Spacecraft Power Lines

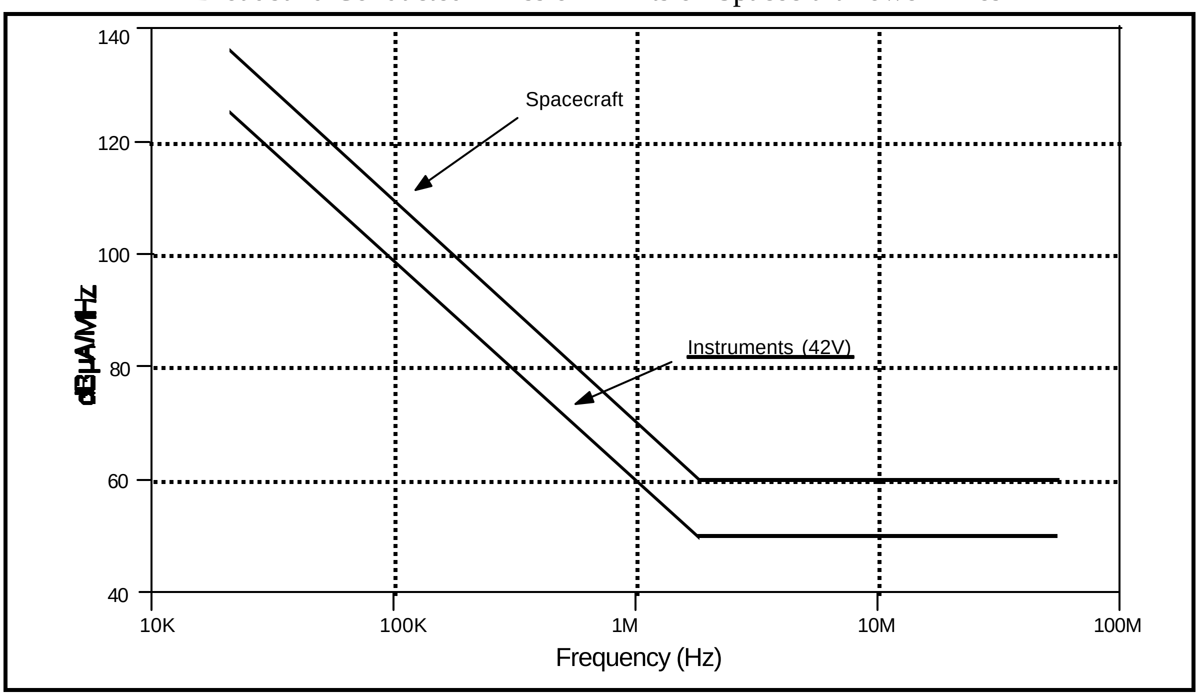


Figure 8.4.5.3
Common Mode Conducted Emission Limits on Primary Instrument (42v) Power Lines

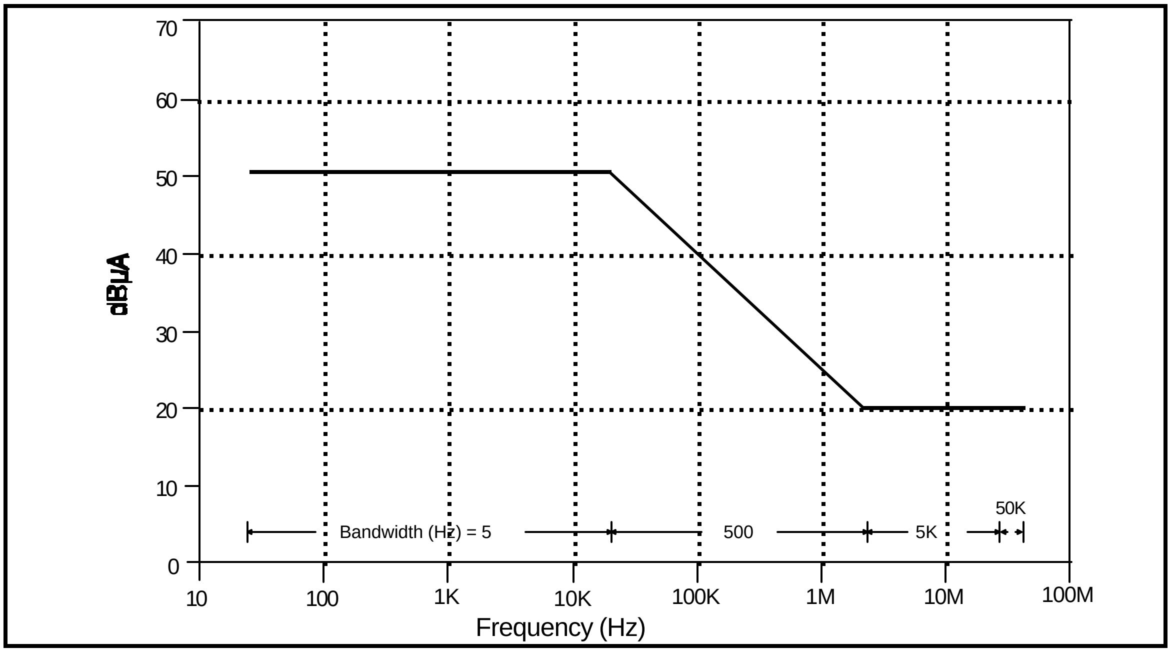
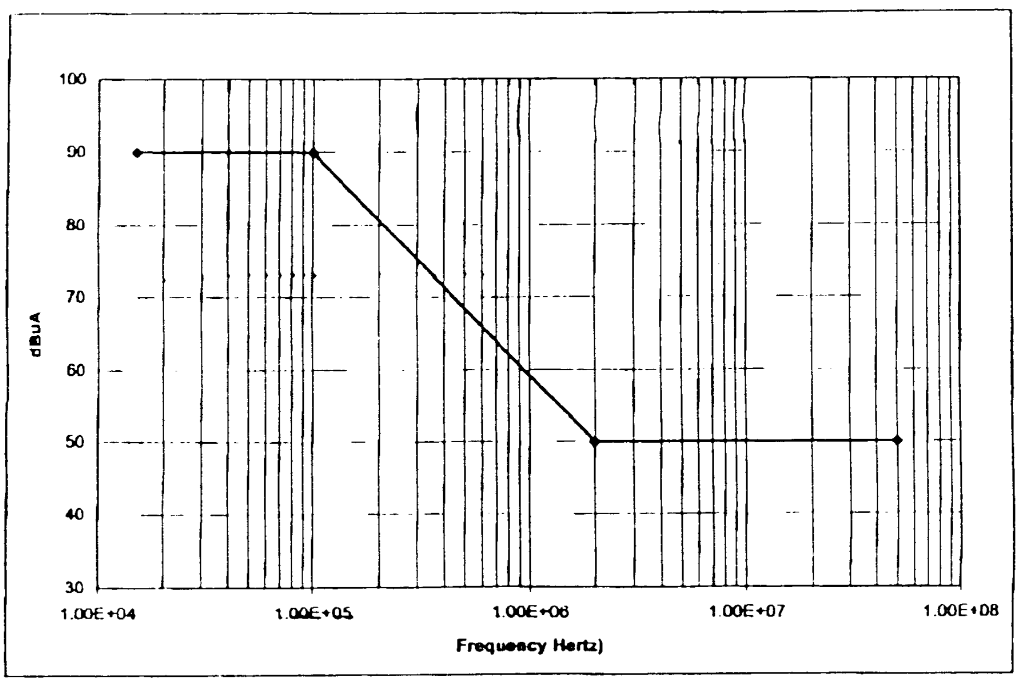


Figure 8.4.5.5-1
Conducted Emission Limits for the Components in the 30v and 53v Busses



8.4.5.4 ***Conducted Emissions on Antenna Terminals*** - Conducted emissions on the antenna terminals of spacecraft receivers and transmitters in key-up modes shall not exceed 34 dBFV for narrowband emissions and 40 dBFV/MHz for broadband emissions.

Harmonics greater than the third and all other spurious emissions from transmitters in the key-down mode shall have peak powers 80 dB down from the power at the fundamental. Power at the second and third harmonics shall be suppressed by 50 + 10 Log [peak power in watts at the fundamental] dB, or 80 dB, whichever requires less suppression.

Testing shall be in accordance with MIL-STD-462, test number CE06. The test is conducted on receivers and transmitters before they are integrated with their antenna systems.

8.4.5.5 ***Conducted Emissions for 30V and 53V Components*** - The conducted emission requirements for components that are connected to the 30V and 53V busses are described in Figure 8.4.5.5-1.

8.4.6 ***Conducted Susceptibility Requirements***

8.4.6.1 ***Conducted Susceptibility CS01 & 02 (Power Lines)*** - This test shall be conducted for instruments connected to the 42V bus over the frequency range of 30 Hz to 400 MHz in accordance with the limit requirements and test procedures of MIL-STD-461C and 462. If degraded performance is observed, the signal level shall be decreased to determine the threshold of interference. Above 50 kHz, modulation of the applied susceptibility signal is required. If the modulation has not been established by component design or mission application, the following guidelines for selecting an appropriate modulation apply:

1. AM Receivers - modulate 50% with 1000 Hz tone.
2. FM Receivers - while monitoring signal-to-noise ratio, modulate with 1000 Hz signal using 10 kHz deviation. When testing for receiver quieting, use no modulation.
3. Components with video channels other than receivers. Modulate 90 to 100% with a pulse of duration 2/BW and repetition rate equal to BW/100, where BW is the video bandwidth.
4. Digital components use pulse modulation with pulse duration and repetition rate equal to that used in the component under test.
5. Non-tuned components use 1000 Hz tone for amplitude modulation of 50%.

8.4.6.1.1 ***Conducted Susceptibility for 30 and 53V Components*** - The conducted susceptibility requirements for components that are connected to the 30V an 53V busses are described in Figure 8.4.6.1.1-1.

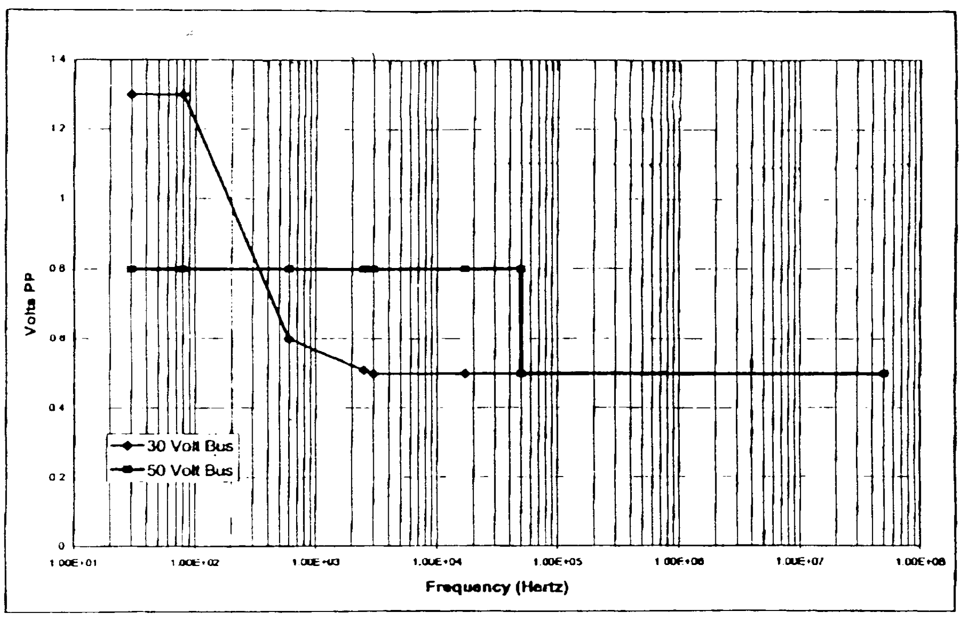
8.4.6.2 ***Conducted Susceptibility CS03 (Two-signal Intermodulation)*** - This test, which determines the presence of intermodulation products from two signals, shall be conducted on receivers operating in the frequency range of 30 Hz to 18 GHz, where this test is appropriate for that type of receiver. The items shall perform in accordance with the limit requirements and the test procedure in MIL-STD-461C and 462 except that the operational frequency range of equipment subject to this test shall be increased to 18 GHz and the highest frequency used in the test procedure increased to 40 GHz.

8.4.6.3 ***Conducted Susceptibility CS04 (Rejection of Undesired Signals)*** - Receivers operating in the frequency range from 30 Hz to 18 GHz shall be tested for rejection of spurious signals where this test is appropriate for that type of receiver. The items should perform in accordance with the limit requirements and the test procedures of MIL-STD-461C and 462 except that the frequency range shall be increased to 40 GHz.

8.4.6.4 ***Conducted Susceptibility CS05 (Cross Modulation)*** - Receivers and tuned amplifiers operating in the frequency range of 30 Hz to 18 GHz shall be tested to determine the presence of cross-modulation products. The items shall perform in accordance with the limit requirements and test procedures of MIL-STD-461C and 462; except that the operational frequency range of equipment subject to this test shall be increased to 18 GHz, and the highest frequency used in the test procedure shall be increased to 40 GHz.

8.4.6.5 ***Conducted Susceptibility CS06 (Powering Transient)*** - A transient signal shall be applied to instrument 42V power lines in accordance with MIL-STD-461C and 462 procedures. The applied transient signal shall equal the powering voltage, with the resulting total voltage at twice the powering level. The transient shall be applied to the input power leads for a duration of 5 minutes at a repetition rate of 60 pulses per second (pps).

Figure 8.4.6.1.1-1
Conducted Susceptibility Limits for the Components in the 30v and 53v Busses



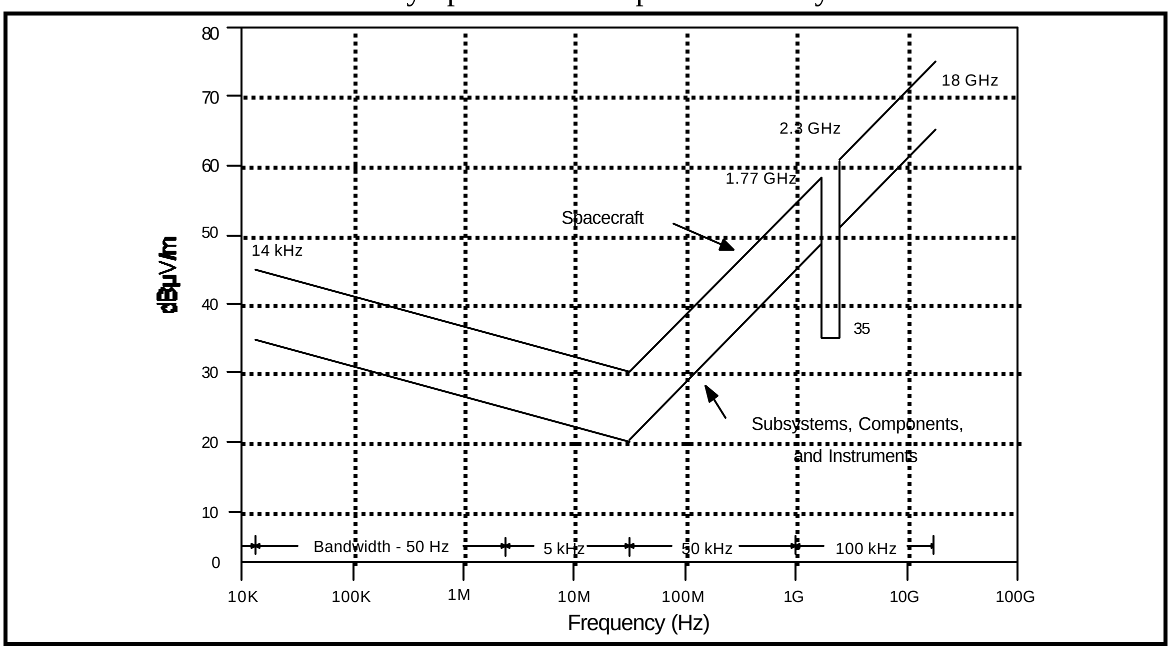
8.4.7 ***Radiated Emission Limits***

8.4.7.1 ***Radiated AC Magnetic Fields*** **-** The radiated AC magnetic field levels produced by the spacecraft shall be limited to 60 dB above 1 pico-Tesla from 20 Hz to 50 kHz. Testing shall be in accordance with MIL-STD-462, test number RE04, with the field measured at a distance of one meter.

8.4.7.2 ***Radiated Electric Fields*** **-** The radiated electric fields shall not exceed the values described below.

8.4.7.2.1 ***Radiated Narrowband Electric Fields*** **-** Unintentional radiated narrowband electric field levels produced by the spacecraft shall not exceed the levels specified in Figure 8.4.7.2.1. Testing shall be in accordance with MIL-STD-461C and 462, test number RE02, with the test frequency range and limits revised as defined in Figure 8.4.7.2.1.

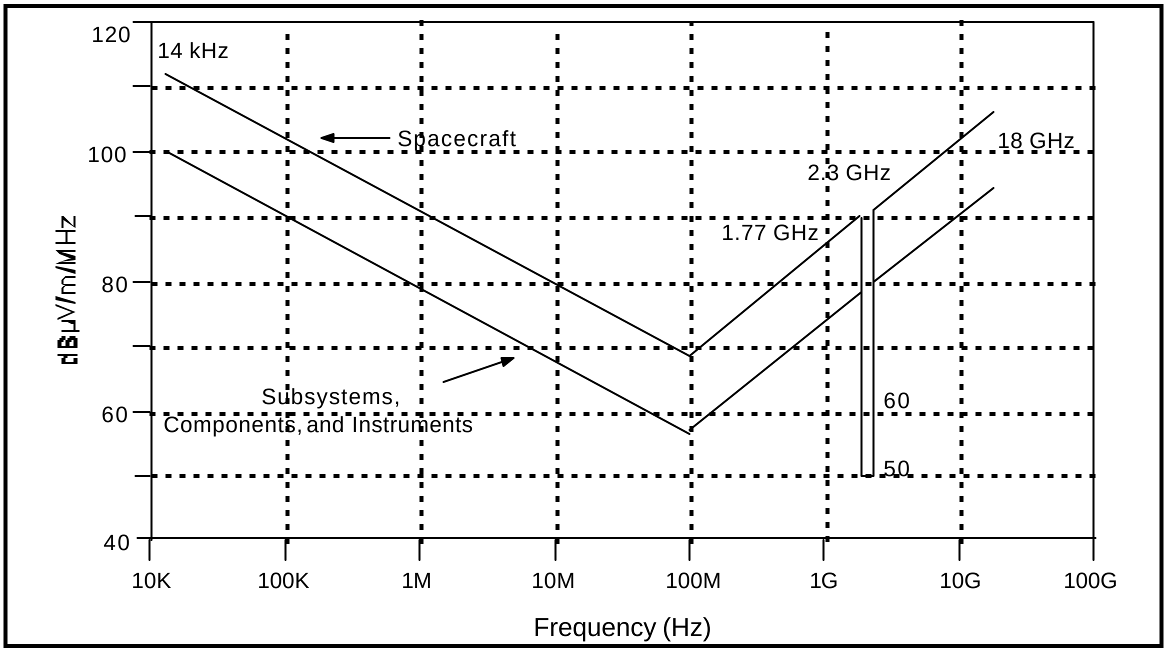
Figure 8.4.7.2.1
Unintentional Radiated Narrowband Limits for E-Field Emission
Produced by Spacecraft and Spacecraft Subsystems



8.4.7.2.2 ***Radiated Broadband Electric Fields*** **-** Unintentional radiated broadband electrical field levels produced by the spacecraft shall not exceed the levels specified in Figure 8.4.7.2.2. Testing shall be in accordance with MIL-STD-461C and 462, test number RE02, with the test frequency range and the limits revised as defined in Figure 8.4.7.2.2. This requirement applies to all continuous emissions. Transient or pulse type noise shall be considered on a case-by-case basis, but shall not in any way detrimentally affect the normal operation of the communications and telemetry and command subsystems, and instruments.

8.4.7.2.3 ***SAR and DCS Receiver Band Emissions*** **-** Unintentional radiated emission measurements in the SAR and DCS receiver bands shall be made in accordance with MIL-STD-461 and 462 RE02 with the EMI meter replaced by a spectrum analyzer preceded by a low noise pre-amplifier such that the test system noise figure is #3 dB. This can be realized with an HP-8566 spectrum analyzer preceded by a MITEQ AU-2A-0550. The instrument under test and associated clock and control signals shall have power applied and the difference in the

Figure 8.4.7.2.2
Unintentional Radiated Broadband Limits for E-Field Emission
Produced by Spacecraft and Spacecraft Subsystems



spectrum analyzer levels shall be noted for both white noise and spurious signals. The test antenna shall be tuned to the center of each of the two frequency bands specified in Table 8.4.7.2.3. Prior to making the actual measurements, the test antenna shall be demated and the cable terminated with 50 ohms. The noise floor of the measuring equipment shall be verified to be lower than the specified maximum signal level in a 100 Hz resolution bandwidth. The specified maximum signal level for all discrete signals and noise power shall be as listed in Table 8.4.7.2.3. The results of this test shall be provided with sufficient sensitivity and resolution to demonstrate that these requirements are met.

This requirement applies to all continuous emissions. Transient or pulse type noise shall be considered on a case-by-case basis, but shall not in any way detrimentally affect the normal operation of the SAR or DCS subsystems.

Table 8.4.7.2.3
SAR and DCS EMI Test Parameters

SAR Band (MHz)	DCS Band (MHz)	Max. Signal Level (dBm)
406.000 to 406.100	401.700 to 402.400	-140

8.4.7.2.4 ***Radiated Spurious and Harmonic Emissions*** - Radiated spurious and harmonic emissions from spacecraft transmit antennas shall have peak powers 80 dB down from the power at the fundamental for harmonics greater than the third. Power at the second and third harmonics shall be suppressed by 50 + 10 Log [peak power in watts at the fundamental] dB, or 80 dB, whichever requires less suppression. These are the same limits as those for conducted spurious and harmonic emissions on antenna terminals.

When MIL-STD-462 and test CE06 for conducted emissions on antenna terminals cannot be applied, test RE03 for radiated spurious and harmonic emissions shall be used as an alternative. Refer to MIL-STD-461C and 462 for details.

8.4.7.2.5 ***Radiated Emissions in a Launch Environment*** - Any unique requirements of or emissions from launch vehicle and launch site transmitters powered on during launch shall be met.

8.4.8 ***Radiated Susceptibility Requirements*** - These tests are based on MIL-STD-461C and 462, as supplemented.

8.4.8.1 ***Radiated Susceptibility Test RS03 (E-field)*** - The spacecraft shall be exposed to external electromagnetic signals in accordance with the requirements and test methods of test RS03. The test shall demonstrate the spacecraft can meet its performance objectives while exposed to the specified levels. Modulation of the applied susceptibility signal is required. If the appropriate modulation has not been established by hardware design or mission scenario, then 50% amplitude modulation by a 100 Hz square wave shall be considered. When performing additional testing at discrete frequencies of known emitters, the emitter modulation characteristics shall be simulated as closely as possible.

1. 2 V/m over the frequency range of 14 kHz to 2 GHz.
2. 5 V/m over the frequency range of 2 to 12 GHz.

The EMI test levels and frequency range shall be increased if it is determined that onboard telemetry systems or other signals in space, such as ground based radars (which are known to produce signals in space in excess of 2 V/m at frequencies at least as low as 400 MHz), could expose the spacecraft to higher levels than the above test levels.

8.4.9 ***Electrostatic Sensitivity Verification Requirement*** - An electrostatic sensitivity susceptibility test or analysis shall be performed on hardware as part of the qualification plan. Hardware previously qualified need not be requalified for GOES N-Q. The test setup is defined in MIL-STD-1541A.

8.4.10 ***RF Airlink Test*** - The spacecraft contractor shall establish a simulated on-orbit RF environment to determine subsystem electromagnetic compatibility with the spacecraft in an on-orbit configuration. The spacecraft antennas shall be connected for free space radiation and shall receive uplink signals. The communication subsystem transmission channels shall be accessed with mission signals described in section 10.2

and its subsections. The spacecraft subsystems shall be configured for on-orbit operation and undergo parametric monitoring for electromagnetic susceptibility. The communication subsystem channels shall be compliant with the spurious requirements in section 10.2.1.12 over all spacecraft subsystem on-orbit configurations. The spacecraft power bus shall also be monitored for interference susceptibility levels.

8.4.11 ***Common Mode Noise* -** Common mode noise shall be less than 0.50 volts peak-to-peak when measured between the power line and chassis, and shall be less than 0.10 volts peak-to-peak when measured between the signal lines and chassis. The measurement shall be performed on all components, including GFE instruments, during their integration with the spacecraft.

8.5 Vacuum, Thermal, and Humidity Verification Requirements

8.5.1 ***General Requirements*** - The vacuum, thermal, and humidity requirements herein apply to the GOES N-Q spacecraft. An appropriate set of tests and analyses shall demonstrate the following capabilities.

1. The spacecraft shall perform satisfactorily within the vacuum and thermal limits.
2. The thermal control system shall maintain the spacecraft and instrument hardware within the mission allowable temperature (MAT) limits during all mission phases.
3. The hardware shall withstand, as necessary, the temperature and humidity conditions of transportation and storage.
4. Hardware workmanship and materials quality shall be sufficient to pass thermal cycle test screening in vacuum and under ambient pressure, as appropriate.

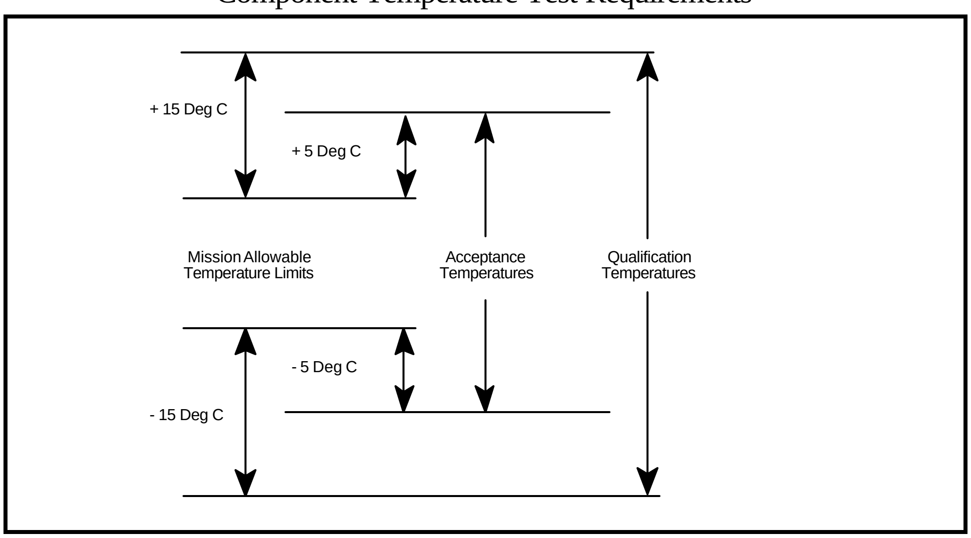
8.5.2 ***Summary of Requirements*** - Table 8.5.3.1 summarizes the tests and analyses that collectively will fulfill the general requirements of section 8.5.

The spacecraft level qualification and acceptance thermal-vacuum verification programs are the same except that a 15°C temperature margin below and above the MAT range is added in the qualification thermal-vacuum test, whereas a 5^{o} temperature margin below and above the MAT range is added in the acceptance thermal vacuum tests.

8.5.2.1 ***Component Temperature Test Requirements*** - The component temperatures during component acceptance tests shall be at least 5°C more extreme than the component MAT limits. The component temperatures during component qualification tests shall be at least 15°C more extreme than the component MAT limits. The component temperature test requirements are illustrated in Figure 8.5.2.1. Equivalent and redundant heater power margin can be substituted for the low temperature margin of 5°C and 15°C for heater controlled components. Components previously qualified to requirements in excess of the GOES N-Q requirements need not be requalified for GOES N-Q.

(Note: Refer to CCR6015A (Deviation Request) in Appendix A)

Figure 8.5.2.1
Component Temperature Test Requirements



8.5.3 ***Thermal-Vacuum Qualification and Acceptance*** **-** The thermal-vacuum program shall ensure that the spacecraft operates satisfactorily in a simulated space environment more severe than expected during the mission.

8.5.3.1 ***Applicability*** - All flight hardware shall be subjected to thermal-vacuum testing in order to demonstrate satisfactory operation in modes representative of mission functions at the MAT, at temperatures in excess of the MAT, and during temperature transitions. The tests shall demonstrate satisfactory operation over the range of possible flight voltages. In addition, hot and cold turn-on shall be demonstrated where applicable.

Spare components shall undergo a test program in which the number of thermal cycles is equivalent to the total number of cycles other flight components are subjected to at the component, subsystem, and spacecraft levels of assembly. Redundant components shall be exercised sufficiently during the test program, including cold and hot starts, to verify proper orbital operations. Testing to validate cross-strapping shall also be performed if applicable.

Table 8.5.3.1
Vacuum, Thermal, and Humidity Requirements

Requirement	**Spacecraft Level**	**Component Level**
Thermal-Vacuum	T	T[1]
Thermal Balance	T, A	T, A
Temperature-Humidity (Transportation & Storage)	A	A
Leakage[2]	T	T

1 Temperature cycling at ambient pressure may be substituted for thermal-vacuum temperature cycling if it can be shown analytically to be acceptable.

2 Hardware that passes this test at a lower level of assembly need not be retested at a higher level unless there is reason to suspect its integrity.

T = Test required.
A = Analysis required; tests may be required to substantiate the analysis.
T, A = Test is not required at all levels of assembly if analysis verification is established for non-tested elements.

Note: Card level thermal analysis is required to insure temperature limits (e.g., junction temperatures) are not exceeded.

8.5.3.2 ***Test Parameters*** **-** The following defines key environmental test parameters:

1. Temperature Cycling - Cycling between acceptance temperature extremes or qualification temperature extremes has the purpose of checking performance at other than stabilized conditions. The minimum number of thermal-vacuum temperature cycles at the spacecraft level and at the component level are as follows:

a. Spacecraft Level - Four thermal-vacuum temperature cycles shall be performed at the spacecraft assembly level. During the cycling, the hardware shall be operating and its performance monitored.
b. Component Level (including SEM instruments) - All hardware shall be subjected to a minimum of eight thermal-vacuum temperature cycles before being installed on the spacecraft; these may include test cycles performed at the subsystem level of assembly. During the cycling, the hardware shall be operating and its performance monitored. For components that are determined by analysis to be insensitive to vacuum effects relative to temperature levels and temperature gradients, the requirements may be satisfied by temperature cycling at normal room pressure in an air or gaseous-nitrogen environment. If this approach is used, the cycling at ambient pressure shall be increased (both the temperature range and the number of cycles) to account for possible analytical uncertainties and to heighten the probability of detecting workmanship defects. The qualification or acceptance temperature margin should be increased by ±5°C if testing at ambient pressure is performed. The number of thermal cycles shall be increased by 50% if testing at ambient pressure.

(Note: Refer to CCR6015A (Deviation Request) in Appendix A)

2. Duration - The total test duration shall be sufficient to demonstrate performance and uncover early failures. The intent of performance testing at each extreme is to ensure that key electrical parameters remain within acceptable limits and do not degrade. Minimum temperature dwell times are as follows:
   a. Spacecraft Level - A minimum of 24 hours at each extreme of each temperature cycle. The thermal soaks shall be of sufficient duration to allow time for performance tests.
   b. Component Level (including SEM instruments) - A minimum of four hours at each extreme of each temperature cycle. The thermal soaks shall be of sufficient duration to allow time for performance tests.

(Note: Refer to CCR6015A (Deviation Request) in Appendix A)

3. Functional Test - Because of the length of time involved, it may be impractical to conduct a comprehensive electrical functional test during spacecraft level thermal-vacuum verification. With GSFC approval, a limited functional test may be substituted if satisfactory performance is demonstrated for the major mission-critical modes of operation.
4. Pressure - The chamber pressure after the electrical discharge checks are conducted shall be less than 1.33 x 10^{-3} Pa (1 x 10^{-5} torr).
5. Turn-on Demonstration - Turn-on capability shall be demonstrated under vacuum at least twice at both the low and high temperatures, as applicable. The ability to function through the voltage breakdown region shall be demonstrated if applicable to mission requirements (all elements that are operational during launch).

(Note: Refer to CCR Number 4102 (Deviation request)in Appendix A)

8.5.3.3 ***Test Setup*** - The setup for the test, including any instrument simulators, shall ensure that the test objectives will be achieved, and that no test-induced problems are introduced. The spacecraft shall be, as nearly as practicable, in flight configuration. Critical temperatures shall be monitored throughout the test and “alarmed” if possible. The spacecraft operational modes shall be monitored.

(Note: Refer to CCR Number 4102 (Deviation request)in Appendix A)

8.5.3.4 ***Demonstration***

1. Electrical Discharge Check - Items that are electrically operational during pressure transitions shall undergo an electrical discharge check to ensure that they will not be permanently damaged from electrical discharge during the ascent and early orbital phases of the mission. The test shall include checks for electrical discharge during the corresponding phases of the vacuum chamber operations.
2. Outgassing Phase - If the test article is contamination sensitive (or if required by the contamination control plan),an outgassing phase shall be included to permit a large portion of the volatile contaminants to be removed. The outgassing phase shall be incorporated into a hot exposure during thermal-vacuum testing. The test item shall be cycled hot and remain at this temperature until the contamination control monitors indicate the outgassing has decreased to an acceptable level.
3. Hot and Cold Start Demonstrations - Start-up capability shall be demonstrated to verify that the test item will turn on after exposure to the extreme temperatures that may occur in orbit. For this check, the test item may be in one of three modes: commanded-off, undervoltage-recycle, or high-voltage.
4. Hot and Cold Conditions - The duration of the hot or cold phase shall be at least sufficient to permit the performance of the functional tests with a minimum soak time of 4 hours for components, 12 hours for subsystems and instruments, and 24 hours for spacecraft testing.
5. Transitions - The test item shall remain in an operational mode during the transitions between temperatures, so its operation can be monitored under the changing environment. The requirement may be suspended when item turn-on is to be demonstrated after a particular transition. In certain cases, it may be allowable to remove thermal insulation to expedite cool-down rates. Caution must be taken, however, not to violate temperature limits or induce test failures through excessive gradients.

8.5.3.5 ***Special Tests*** - Special tests are required to evaluate unique features, such as a radiation cooler, or to demonstrate the performance of external devices, such as solar array hinges or deployable booms, that are deployed after the spacecraft has attained orbit. The test configuration shall reflect, as nearly as practicable, the configuration expected in flight. In particular, a temperature/illumination test shall be performed on the assembled solar array at the upper operating temperature to demonstrate adequate power capacity.

8.5.3.6 ***Trouble-free Performance*** - At least 100 trouble-free hours of functional operations shall be demonstrated in the thermal verification program (refer to section 8.1.4).

8.5.4 ***Thermal Design Verification*** - Verification of the thermal control requirements shall be demonstrated through successful spacecraft level thermal balance test results, successful pre-flight temperature predictions, and successful flight temperatures.

The spacecraft level thermal balance test program shall demonstrate successful thermal performance under the conditions of: (1) summer solstice, (2) winter solstice, (3) equinox, and (4) equinox eclipse, starting from condition 3 steady state. The thermal balance test program will be considered successful if the combined comprehensive spacecraft-instrument analytical model can correlate predicted temperatures with test data, and if: (1) component or simulated component temperatures are all within the component MAT limits (operating and non-operating, as appropriate), and (2) all transistor collector junction temperatures are as specified in section 10.6.2.

In addition to successful spacecraft level thermal balance test results, thermal control requirements shall also be verified by demonstrating combined comprehensive spacecraft-instrument analytical model preflight predicted flight temperatures within the MAT limits, and by achieving actual flight temperatures within the MAT limits.

8.5.5 ***Leakage (Integrity Verification)*** - Tests shall be conducted on sealed items to determine whether leakage exceeds the rate prescribed for the mission.

8.5.5.1 ***Levels of Assembly*** - Tests shall be conducted on the component and spacecraft levels of assembly.

8.5.5.2 ***Demonstration*** - Leakage rates shall be checked before and after stress-inducing portions of the verification program. The final check shall be conducted after all environmental testing. A mass spectrometer may be used to detect flow out of or into sealed items.

If dynamic seals are used, the item shall be operated during the test, otherwise operation is not required. The test shall be conducted under steady-state conditions, i.e., stable pumping, pressures, temperatures, etc. If time constraints do not permit the imposition of such conditions, a special test method shall be devised.

8.6 Structural and Mechanical Verification Requirements

8.6.1 ***General Requirements*** - Table 8.6.1 summarizes the structural and mechanical verification activities discussed in this section. When the tests and analyses are planned, consideration shall be given to expected structural loads, vibroacoustics, sine vibration, mechanical shock, and pressure profiles induced during all phases of the mission.

The verification test must envelope the expected mission levels, with appropriate margins added for qualification, and impose sufficient stress to detect workmanship faults.

Structural load tests of some components shall be necessary if they cannot be properly applied during testing at higher levels of assembly. Ground handling, transportation, and test fixtures shall be analyzed and tested for proper strength as required by safety, and shall be verified for stability for applicable configurations as appropriate.

A protoflight approach shall be used for structural and mechanical verification. That is, GOES-N shall be tested to qualification levels for acceptance durations. GOES-O,P,Q shall be tested to acceptance levels for acceptance durations.

Table 8.6.1
Structural and Mechanical Verification Test Requirements

Requirement	Spacecraft	Components
Structural Loads		
Modal Survey	A, T	-
Design Qualification (see 10.7)	A, T/A	A, T/A
Structural Reliability (see 10.7)		
Primary & Secondary Structure	A, T[1]	-
Vibroacoustics		
Acoustics	T	-
Random Vibration		T[1]
Sine Vibration	T	T[1]
Mechanical Shock	T	T[1]
Mechanical Function	A, T	-
Pressure Profile	-	A
Mass Properties	A, T	A, T[1]

A = Analysis required.
T = Test required.
A, T/A = Analysis and Test or analysis only if no-test safety factors given in section 10.7 are used.
T[1] = Test shall be performed unless assessment justifies deletion.

8.6.2 ***Modal Survey*** - A modal survey test to verify that the analytical model adequately represents the hardware's dynamic behavior is required for the spacecraft and subsystems that do not meet the minimum fundamental frequency requirement. The minimum fundamental frequency, in turn, is dependent on the launch vehicle. A determination will be made on a case-by-case basis and shall be specified in the design and test requirements. A low level sine survey is generally an appropriate method for determining the fundamental frequency.

The results of the modal survey are required to identify any inaccuracies in the mathematical model used in the coupled loads analysis and structural analysis programs, so that modifications can be made if needed. Such experimental verification is required because a degree of uncertainty exists in unverified models, thereby reducing confidence in the accuracy of the set of limit loads derived therefrom.

If a modal survey test is required, all significant modes up to the required frequency must be determined both in terms of frequency and mode shape. Any test method that is capable of meeting the test objectives with the necessary accuracy may be used to perform the modal survey. The input forcing function may be transient, fixed frequency, swept sine wave, or random in nature. Requirements for correlating model results with the test data are given in section 3.2.2 of the GOES N-Q Statement of Work S-415-23.

When a satisfactory modal survey has been conducted on a representative structural model, a modal survey of the protoflight unit may not be necessary. A representative structural model is defined as one that duplicates the structure as to materials, configuration, fabrication, and assembly methods, and satisfactorily simulates other items that mount on the structure as to location, method of attachment, weight, mass properties, and dynamic characteristics.

8.6.3 ***Vibroacoustic Qualification*** **-** For the vibroacoustic environment, qualification levels for prototflight hardware are defined as the flight limit level plus 3 dB. Protoflight test durations are one minute per axis. When random vibration levels are determined, responses to the acoustic inputs plus the effects of vibration transmitted through the structure shall be considered. As a minimum, the vibroacoustic test levels shall be sufficient to demonstrate acceptable workmanship.

For all vibroacoustic testing, the test item shall be in an operational configuration, both electrically and mechanically, representative of its configuration at lift-off. Deviations from this requirement may be allowed if justified by analysis and/or test. During the test, proper performance shall be monitored. Before and after test exposure, the test item shall be examined and functionally tested.

8.6.3.1 ***Spacecraft Level Acoustic Test*** **-** The fully integrated spacecraft, including flight instruments, shall be subjected to an acoustic test in a reverberant sound pressure field to verify its ability to survive the lift-off acoustic environment and to provide a final workmanship acoustic test. If the test specification derived from the launch vehicle expected environment, including fill-factor, is less than 138 dB, the test profile shall be raised to provide a 138 dB test level. A spacecraft level random test is not required.

8.6.3.2 ***Component Level Random Vibration Tests*** **-** As a screen for design and workmanship defects, components, including the SEM instruments, shall be subjected to a random vibration test along each of three mutually perpendicular axes. In addition, an acoustic test shall be performed if any components are particularly sensitive to the acoustic environment.

The component random vibration spectrum shall be based on levels measured at the component mounting locations during previous subsystem or spacecraft testing. When such measurements are not available, the levels shall be based on statistically estimated responses of similar components on similar structures or on spacecraft analysis. In the absence of any knowledge of the expected levels, the generalized vibration test specification of Table 8.6.3.2-1 may be used. As a minimum, all components shall be subjected to the levels of Table 8.6.3.2-2.

(Note: Refer to CCR Number 4102A and 4103 (Deviation request)in Appendix A)

8.6.3.3 ***Acceptance Requirements*** **-** Vibroacoustic testing for the acceptance of previously qualified hardware shall be conducted at flight limit levels using the same duration as recommended for protoflight hardware. As a minimum, the acoustic test level shall be 138 dB, and the random vibration levels shall represent the workmanship test levels defined in Table 8.6.3.2-2.

The spacecraft shall be subjected to an acoustic test. The components and SEM instruments shall be subjected to random vibration tests in three axes. The components and SEM instruments shall be integrated prior to the spacecraft level acoustic test.

(Note: Refer to CCR6015A (Deviation Request) in Appendix A)

Table 8.6.3.2-1
Generalized Random Vibration Test Levels Components 22.7 kg (50 lb) or Less

Frequency (Hz)	**ASD Level (g^2/Hz)**	
	Qualification	**Acceptance**
20	0.026	0.013
20-50	+6 dB/oct	+6 dB/oct
50-800	0.16	0.08
800-2000	-6 dB/oct	-6 dB/oct
2000	0.026	0.013
Overall	14.1 g_{RMS}	10 g_{RMS}

The acceleration spectral density level may be reduced for components weighing more than 22.7 kg (50 lb) according to:

		Weight in kg	Weight in lb	
dB reduction	=	10 log (W/22.7)	10 log (W/50)	
$ASD_{(50\text{-}800\ Hz)}$	=	0.16•(22.7/W)	0.16•(50/W)	for protoflight
$ASD_{(50\text{-}800\ Hz)}$	=	0.08•(22.7/W)	0.08•(50/W)	for acceptance

where W = component weight.

The slopes shall be maintained at ±6 dB/oct for components weighing up to 59 kg (130 lb). Above that weight, the slopes shall be adjusted to maintain an ASD level of 0.01 g^2/Hz at 20 and 2000 Hz.

For components weighing over 182 kg (400 lb), the test specification will be maintained at the level for 182 kg (400 lb).

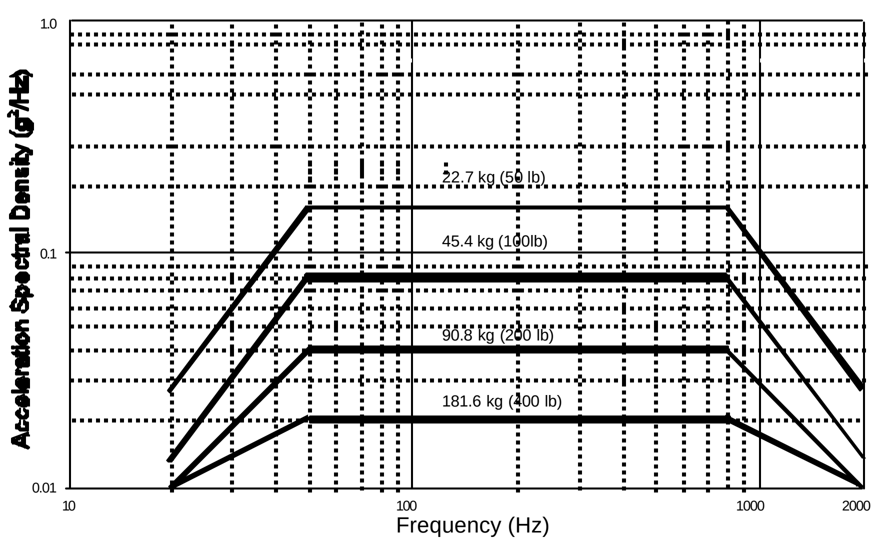


Table 8.6.3.2-2
Minimum Workmanship Random Vibration Test Levels
Components 45.4 kg (100 lb) or Less

Frequency (Hz)	**ASD Level (g^2/Hz)**
20	0.01
20-80	+3 dB/oct
80-500	0.04
500-2000	-3 dB/oct
2000	0.01
Overall	6.8 g_{RMS}

The plateau acceleration spectral density level (ASD) may be reduced for components weighing between 45.4 and 182 kg, or 100 and 400 pounds according to the component weight (W) up to a maximum of 6 dB as follows:

		Weight in kg	Weight in lb
dB reduction	=	10 log(W/45.4)	10 log(W/100)
$ASD_{(plateau)}$ level	=	0.04•(45.4/W)	0.04•(100/W)

The sloped portions of the spectrum shall be maintained at plus and minus 3 dB/oct. Therefore, the lower and upper break points, or frequencies at the ends of the plateau become:

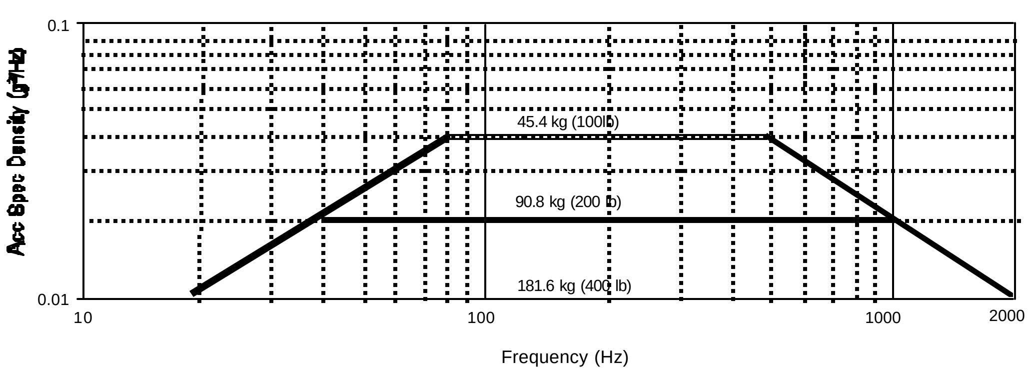
F_L = 80 (45.4/W) [kg]
= 80 (100/W) [lb]

F_L = frequency break point low end of plateau

F_H = 500 (W/45.4) [kg]
= 500 (W/100) [lb]

F_H = frequency break point high end of plateau

The test spectrum shall not go below 0.01 g^2/Hz. For components whose weight is greater than 182 kg or 400 lb, the workmanship test spectrum is 0.01 g^2/Hz from 20 to 2000 Hz with an overall level of 4.4 g_{RMS}.



(Note: Refer to CCR Number 4102A and 4103 (Deviation Request) in Appendix A)

8.6.4 ***Sinusoidal Sweep Vibration Qualification*** - Qualification for the low frequency sine transient or sustained sine environments requires swept sine vibration tests at the spacecraft and component levels of assembly. The qualification level for the sinusoidal vibration environment is defined as the limit level times 1.25. The test input frequency range shall be limited to the band from 5 to 40 Hz. Protoflight sweep rates are 4 octaves/minute. Test items shall be examined and functionally tested before and after each vibration test, with proper performance monitored during each test.

8.6.4.1 ***Spacecraft-level Sine Sweep Vibration Tests*** - The spacecraft shall be attached to the vibration fixture by use of a flight-type launch vehicle attach fitting (adapter) and attachment (separation system) hardware. The spacecraft shall be in an operational configuration, both electrically and mechanically, representative of its configuration at lift-off. Deviations from this requirement may be allowed if justified by analysis and/or test. Sine sweep vibration shall be applied at the base of the adapter in each of three orthogonal axes, one of which is parallel to the thrust axis. The test shall be performed by sweeping the applied vibration once through the 5 to 40 Hz frequency range in each axis. Mission specific sine sweep test levels shall be developed based on the coupled loads analysis or obtained from the launch vehicle user's guide..

During the sine sweep vibration test, loads induced in the spacecraft and/or adapter structure while sweeping through resonance shall not exceed 1.25 times flight limit loads. If required, test levels shall be reduced (notched) at critical frequencies. Acceleration responses of specific critical items may also be limited to 1.25 times flight limit levels, provided that the spacecraft model used for the coupled loads analysis has sufficient detail and the specific responses are recovered.

8.6.4.2 ***Component-level (including SEM Instruments) Sine Sweep Vibration Tests*** - The test item in its launch configuration shall be attached to the test equipment by a rigid fixture. The mounting shall simulate, insofar as practicable, the actual mounting of the item in the spacecraft, with particular attention given to duplicating the mounting interface. All connections to the test item (connectors, harnesses, etc.) shall be simulated with lengths at least to the first tie-down point. In mating the test item to the fixture, flight-type mounting and fasteners, including torque levels and locking features, shall be used. Sine sweep vibration shall be applied at the base of the test item in each of three mutually perpendicular axes.

Components with test verified first modes greater than 60 Hz may be excluded from this requirement. Risk of exposure to low frequency sine vibration for the first time at the spacecraft-level test should be considered before deleting component level sine vibration.

8.6.4.3 ***Acceptance Requirements*** - Sine sweep vibration testing for the acceptance of previously qualified hardware shall be conducted at the flight limit levels using the same sweep rates as used for protoflight hardware.

Components with test verified first modes greater than 60 Hz may be excluded from this requirement. Risk of exposure to low frequency sine vibration for the first time at the spacecraft-level test should be considered before deleting component-level sine vibration.

(Note: Refer to CCR6015A (Deviation Request) in Appendix A)

8.6.5 ***Mechanical Shock Qualification*** - Both self-induced and externally induced shocks shall be considered in defining the mechanical shock environment. For all mechanical shock tests, the test item shall be in the electrical and mechanical operational modes appropriate to the mission phase in which the shock will occur. The test item shall be examined and functionally tested before and after the mechanical shock tests, and proper performance shall be monitored during each test.

8.6.5.1 ***Component Mechanical Shock Tests*** - All components shall be qualified for the following mechanical shock environments:

1. Self-Induced Shock - The component shall be exposed to self-induced shocks by actuation of all shock-producing devices. Hardware shall be exposed to each shock source twice to account for the scatter associated with the actuation of the same device. The internal spacecraft flight firing circuits shall be used to trigger the event rather than external test firing circuits. This testing may be deferred to the spacecraft level of assembly with NASA concurrence.
2. Externally Induced Shock - Mechanical shocks originating from other components, instruments, or launch vehicle operations shall be assessed. When the severest shock is externally induced, a suitable simulation of that shock shall be applied at the component interface. When it is feasible to apply this shock with a controllable shock-generating device, the qualification level shall be 1.4 times the maximum expected value at the component interface, applied once in each of the three axes.

(Note: Refer to CCR6015A (Deviation Request) in Appendix A)

8.6.5.2 ***Spacecraft-level Mechanical Shock Tests*** - The spacecraft must be qualified for the shock induced during spacecraft separation and for any other externally induced shocks whose levels are not enveloped at the spacecraft interface by the separation shock level. Mechanical shock testing may be performed at the spacecraft level of assembly to satisfy the subsystem mechanical shock requirements of section 8.6.5.1.

If launch vehicle induced shocks or shocks from other sources are not enveloped by the separation test, the spacecraft shall be subjected to a test designed to simulate the greater environment. If a controllable source is used, the qualification level shall be 1.4 times the maximum expected level at the spacecraft interface, applied once in each of the three axes.

8.6.5.3 ***Acceptance Requirements*** - The need to perform mechanical shock tests for the acceptance of previously qualified hardware shall be considered on a case-by-case basis. Testing shall be given careful consideration by evaluating mission reliability goals, shock severity, hardware susceptibility, design changes from the previous qualification configuration including proximity to the shock source, and previous history.

8.6.6 ***Mechanical Function Verification***

8.6.6.1 ***Component Testing*** - Each component that performs a mechanical operation shall undergo functional qualification testing. With NASA concurrence, such testing may be performed at the spacecraft level of assembly. The test is conducted after any other testing that may affect mechanical operation to confirm proper performance and to assure that no degradation has occurred during the previous tests.

During the test, electrical and mechanical components shall be in the appropriate operational mode. The component shall also be exposed to pertinent environmental effects that may occur before and during mechanical operation. For each mechanical operation, such as appendage deployment, tests shall be performed at nominal, low, and high energy levels. The nominal level test shall be conducted at the most probable level that will occur during a normal mission. High and low energy levels shall be conducted to prove positive strength and function margins. The margins shall take into account all the operation, strength, and test uncertainties, including the adverse interaction of potential extremes of temperature, friction, spring forces, etc. Components shall be examined and electrically tested before and after each test, and proper component performance shall be monitored during all tests.

8.6.6.2 ***Spacecraft-level Testing*** - A series of mechanical function tests shall be performed on the spacecraft to demonstrate release, motion, and lock-in of all appendages and other mechanical devices. Unless the design of the device dictates otherwise, mechanical testing may be conducted in ambient laboratory conditions. If any device is removed from the spacecraft, the testing shall be repeated after reinstallation of the device.

8.6.7 ***Pressure Profile Qualification*** - Spacecraft hardware shall be qualified for the pressure profile environment by analysis and/or test. An analysis shall be performed to estimate the pressure differential induced by the nominal launch trajectories across elements susceptible to such loading (i.e., thermal blankets and component housings). If analysis does not indicate a positive margin at loads twice those induced by the maximum expected pressure differential, testing is required.

The flight pressure profile shall be determined by the analytically predicted pressure-time history inside the payload fairing for the mission's nominal launch trajectory. The qualification pressure profile is determined by increasing the predicted flight rate by a factor of 1.12, and shall be applied once. During the test, the unit under test shall be in the electrical and mechanical operational modes that are appropriate for the event being simulated. The unit shall be examined and functionally tested before and after the pressure profile test, and the unit's performance shall be monitored during each test.

Pressure profile test requirements do not apply for the acceptance testing of previously qualified hardware.

8.6.8 ***Mass Properties*** - The mass properties program shall include an analytic assessment of the spacecraft's ability to comply with mission requirements, supplemented as necessary by measurement. The spacecraft mass properties, including weight, center of gravity, moments, and products of inertia, shall be determined by analysis and/or measurement for all spacecraft configurations. Determination of component properties shall be sufficiently accurate that, when combined analytically to derive the mass properties of the spacecraft, the uncertainties shall be small enough to ensure compliance with requirements. If analytic determination of spacecraft mass properties is unfeasible, then direct measurement is required.

8.6.9 ***Life Testing*** - A life test program is required for mechanisms and electromechanical devices. The verification plan shall address the life test program, identifying the elements that require such testing and the test methods to be employed. Hardware previously life tested to requirements in excess of the GOES N-Q requirements need not be re-tested.

The life test mechanism shall be fabricated and assembled such that it is as nearly as possible identical to the actual flight mechanism (prefer flight spare). Prior to the start of life testing, mechanisms shall be subjected to the same ground testing environments that are anticipated for the flight units. The mechanism's thermal environment during the life test shall be representative of the on-orbit environment, including temperature cycling and gradients. Consideration shall be given to including the effects of vacuum on the performance of the mechanism, with particular attention to its effects on the thermal environment and potentially adverse effects on lubrication and materials. Physical parameters that are an indication of the health of the mechanism shall be closely monitored and trended during the life test. Redundant sensors shall be provided for all critical test data.

The test spectrum for the life test shall represent the required mission life for the flight mechanism, including both ground and on-orbit mechanism operations. It shall include, if applicable, a representative range of velocities, duty cycles, number of direction reversals, and number of dead times or stop/start sequences between movements. The recommended minimum goal for the life test is to achieve a 25% margin on mission life. Pre- and post-life test baseline performance tests shall be conducted with clear requirements established for determining minimum acceptable performance at end-of-life.

Upon completion of the life test, the proper level of inspections shall be conducted. Abnormal wear, significant lubrication breakdown, or excessive debris generation may be cause for declaring the life test a failure, despite completion of the required test spectrum. A thorough investigation of all moving components and wear surfaces shall be conducted.

For items determined not to require life testing, the rationale for eliminating the test shall be provided along with a description of the analyses that will be done to verify the validity of the rationale.

8.7 GFE Instrument Testing

Spacecraft environmental testing shall not expose the flight GFE instruments to vibration or thermal limits exceeding those defined in the instrument ICDs.

8.7.1 ***Test Point Access*** **-** The spacecraft shall accommodate GFE instrument testing by providing access to the instrument accelerometers, thermocouples, and test point connectors throughout the environmental test program.

8.7.2 ***Imager and Sounder Testing***

8.7.2.1 ***Cooler Door Deployment*** **-** The spacecraft shall accommodate radiant cooler cleaning and cooler cover door deployment testing by enabling the door hinge axes to be aligned perpendicular to the ground.

8.7.2.2 ***Optical Performance Testing*** **-** The spacecraft shall accommodate Imager and Sounder optical testing by enabling the optical axes of the telescope to be aligned perpendicular to the ground.

9.0 INSTRUMENTS

9.1 Space Environment Monitor Subsystem

The space environment monitor (SEM) subsystem shall provide measurements of the ambient magnetic field vector, solar X-ray flux (two channels from 0.05 to 0.8 nm), solar EUV flux (five channels from 10 to 125 nm), and multiple measurements characterizing the charged particle population, including the electron, proton, and alpha particle fluxes, throughout the 24-hour day. All SEM instruments shall be capable of operating and transmitting data during eclipses, and each instrument shall be capable of independent operation.

9.1.1 ***Magnetic Field Measurement***

9.1.1.1 ***General -*** Data transmitted from the spacecraft shall allow the real-time determination of the magnitude and direction of the ambient magnetic field in an Earth-referenced coordinate system. The spacecraft contractor shall supply all calibration information and algorithms necessary for determination of the ambient field. These include, but are not limited to, sensor offsets and gains. If stray or intentional (e.g., torquer) fields from the spacecraft are added to the ambient field measured at the sensor, data must be transmitted in telemetry to enable the sensor data to be corrected in real time at the SEC; and the spacecraft contractor shall be responsible for providing the correction algorithms required to meet the performance specifications listed below.

9.1.1.2 ***Design Guidance Information*** - The following information is provided for guidance in meeting the specifications listed below. As a goal, spacecraft static and dynamic fields at the sensor location should be eliminated by design to reduce the complexity and frequency of ground data processing corrections. In particular, instantaneous spacecraft field changes due to changes in system states resulting in signatures at the sensor greater than the noise specification should be avoided, because of the transient errors which may result during ground correction. Particular care must be taken to eliminate permeable and permanent magnetic materials close to the sensor which can introduce unknown or unstable offsets to the measurements. A ground test program shall be performed to demonstrate on-orbit specifications will be met. Examples of testing include: spacecraft stray and DC magnetic interference testing and periodic monitoring of instrument offsets and other characteristics.

9.1.1.3 ***Dynamic Range*** - The magnetic field measurement shall be performed within the accuracy specified in section 9.1.1.6 for an ambient field of any direction having a magnitude equal to or less than 400 nano-Tesla (nT).

9.1.1.4 ***Data Sampling Rate*** - Data for determining each component of the ambient magnetic field shall be telemetered at a nominal rate of 2 Hz ±20%. The effective time of each component shall be known in accordance with the provisions of section 10.1.3.2. Each individual component shall be sampled uniformly in time and simultaneously within 25% of the sample period (i.e., within 0.125 seconds for a 2 Hz sampling rate).

9.1.1.5 ***Bandwidth*** - The frequency response of the instrument shall strongly discriminate against the possibility of aliasing of the magnetometer data resulting from the ambient background and spacecraft interference. The nominal measurement bandwidth for each component of the vector field shall be defined by a pre-sampling filter approximating a fifth order Butterworth filter with greater than 25 dB attenuation at the Nyquist folding frequency. The nominal 3 dB attenuation frequency shall be one-half the Nyquist folding frequency, with the three components matched to the nominal 3 dB frequency ±0.75%. The actual frequency response of the measurement shall not deviate from the nominal by more than 0.3 dB in amplitude or 5° in phase anywhere within the nominal 3 dB bandwidth.

9.1.1.6 ***Accuracy*** - The component of the ambient field along each of three orthogonal axes, after ground correction for the spacecraft static and dynamic fields, shall be determined to an absolute accuracy of ±4.0 nT over the instrument bandwidth without correction for temperature effects. The spacecraft shall provide temperature calibrations for reducing the error in this determination to ±1 nT. If the ±1 nT accuracy, as defined above, can be achieved without temperature compensation, temperature compensation factors are unnecessary. Each component shall have a resolution of 0.1 nT or better when telemetered. The three axes shall be orthogonal to within ±0.5°, and the orientation of the axes in the spacecraft coordinate system shall be determined to an accuracy of ±1° and be stable to ±0.25°. Alternatively, non-orthogonality of the raw instrument measurement axes exceeding 0.5 degrees but less than 1.0 degrees may be ground corrected by application of an orthogonalization matrix supplied as part of the ground calibration of each instrument. Such correction shall be applied after any required ground correction for temperature and/or spacecraft static and dynamic fields.

9.1.1.7 ***Noise*** - The estimate of the magnitude of the ambient field computed on the ground along each axis shall not fluctuate by more than 0.3 nT, 3ó, when the spacecraft is in a normal operational mode. Unavoidable step changes, which create out of specification transients after ground correction, may occur. They shall not, however, average more than one transient in any one hour period; and the duration of the out of specification transient shall be no greater than five seconds.

9.1.1.8 ***In-flight Calibration*** - An on-orbit capability shall be provided to demonstrate the basic instrument is functional. This shall be accomplished by adding known fields to the ambient field by ground command. A capability shall be provided to determine from the calibration data the sensitivities necessary for calculating the ambient-plus spacecraft field. This determination shall include calibration field levels covering more than 50% of the full range of the magnetometer. This determination shall also be stable and repeatable during times of quiet ambient field to an accuracy of ±1 nT in each axis at each field level. Magnetometer calibration shall be terminated by ground command. A constraint shall be included in the GTACS schedule constraint utility to ensure a calibration-terminate command is inserted in the command schedule a contractor-specified period of time after a calibration start command. The spacing between these calibration start and end commands shall allow sufficient time to complete the calibration operation and minimize data loss. In addition, magnetometer calibration shall not preclude other spacecraft commanding.

9.1.1.9 ***In-flight Magnetometer Effective Offset Determination*** - The effective magnetometer sensor offset (sensor plus spacecraft) shall be determined on-orbit via a spacecraft rotation maneuver (deleted words). The determination shall be made (deleted word) during the spacecraft post-launch test period in the vicinity of local noon.

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9.1.2 *Solar X-Ray and EUV Sensor*

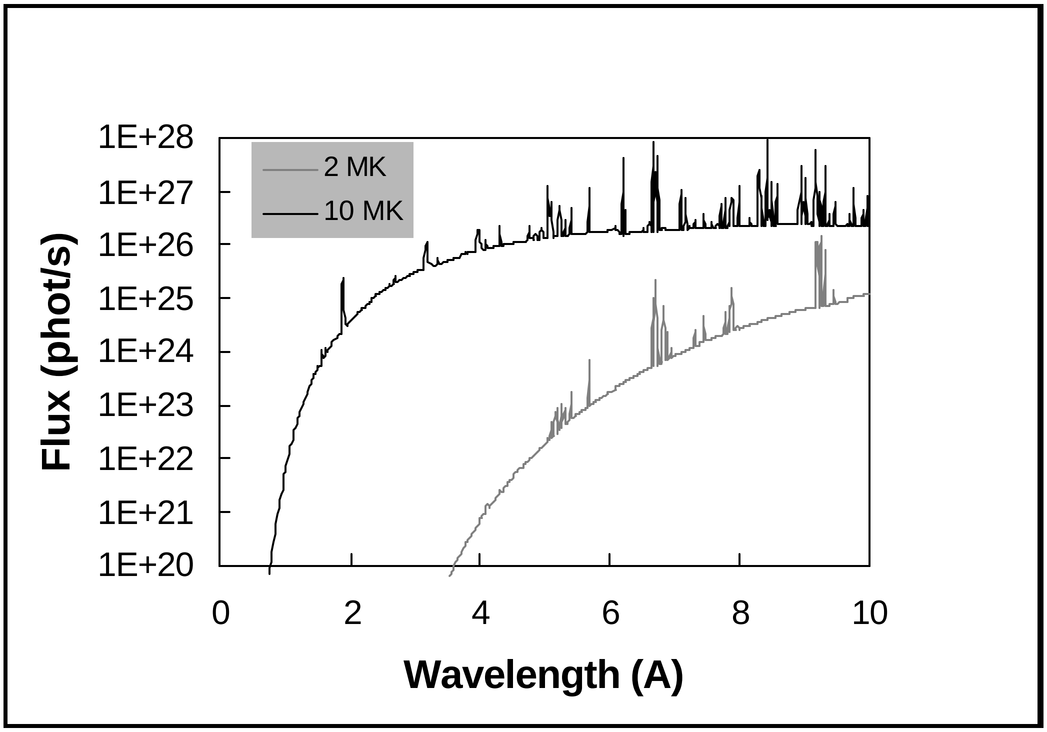
9.1.2.1 ***Wavelength Ranges and Threshold Sensitivities*** - Two X-ray channels are required to observe solar fluxes in the 0.05 to 0.3 nm and the 0.1 to 0.8 nm bands, respectively. At least five EUV channels are required to observe solar fluxes in the 10-125 nm range. The preferred and the acceptable range of values for the lower and upper XRS/EUV channel band edges are provided in Table 9.1.2.1.

The instrument shall be capable of measuring fluxes above a threshold sensitivity for each of the seven channels listed in Table 9.1.2.1. Threshold sensitivity is defined as the minimum in-band solar flux that will produce a mean signal equal to the standard deviation of the data (instrumental noise) over a 10-second interval. For the two X-ray channels, the in-band solar spectrum varies several orders of magnitude with wavelength as shown in Figure 9.1.2.1 (from Mewe and Groenschild, 1981, Astron. Astrophys. Suppl. Ser., vol. 45, pp 11-52). A solar spectrum at 2 x 10^6 K shall be assumed for the two X-ray channels for threshold sensitivity determination. For the EUV channels, the solar spectrum is not a strong function of wavelength and a flat spectrum may be assumed within each channel for determining the threshold sensitivity. The out-of-band rejection shall be such that < 10% of the observed signal comes from out-of-band for a typical solar spectrum. Alternatively, a method of measuring out-of-band contamination shall be provided. Over the duration of the mission, the instrument response shall not change by > 5% due to deposition of outgassed materials onto optical surfaces.

Table 9.1.2.1 <u>CCR6057A, Mod 32</u>
Wavelength Ranges

Channel	**Preferred Wavelength Range (nm)**		**Acceptable Lower Band Edge Range (nm)**	**Acceptable Upper Band Edge Range (nm)**
	Lower	**Upper**		
XRS-A	0.05	0.4	0 to 0.1	0.3 to 0.5
XRS-B	0.1	0.8	0 to 0.2	0.75 to 1.2
EUV-A	10	25	5 to 12	<u>15</u> to 27
EUV-B	25	40	17 to 27	34 to 43
EUV-C	40	65	37 to <u>52</u>	62 to 70
EUV-D	65	100	<u>10</u> to 85	<u>20</u> to 110
EUV-E	119	124	114 to 120	123 to 129

Figure 9.1.2.1
Solar X-ray Spectra for an Emission Measure of 10^{44} cm^{-3}



9.1.2.2 ***Threshold Fluxes and Dynamic Range*** - The instrument shall be capable of measuring fluxes for each channel over the specified dynamic ranges given in Table 9.1.2.2, when the fluxes are above the given thresholds.

Table 9.1.2.2
Required Sensitivities

Channel	Threshold Flux	Min. Dynamic Range
XRS-A	5×10^{-9} W/m^2	10^5
XRS-B	2×10^{-8} W/m^2	10^5
EUV-A	1×10^{-6} W/m^{2}nm*	10^3
EUV-B	2×10^{-6} W/m^2 nm *	10^3
EUV-C	1×10^{-6} W/m^{2}nm*	10^2
EUV-D	2×10^{-6} W/m^{2}nm*	10^2
EUV-E	1×10^{-4} W/m^{2}nm*	10^2

* The threshold flux levels represent values 1/10 the expected minimum solar flux.

9.1.2.3 ***Flux Resolution and Response*** - For the two X-ray channels, the resolution of the energy flux measurements shall be < 2% of the detected flux for fluxes > 20 times the threshold fluxes specified in Table 9.1.2.2. The telemetered data shall be a monotonic function of the input fluxes with deviations from a monotonic response function being less than the flux resolution when the incident flux exceeds 20 times the threshold flux. For the EUV channels the resolution of the energy flux measurement shall be < 0.25% of the full scale value.

9.1.2.4 ***Electron Environment*** - The minimum performance of the Solar X-ray and EUV sensor shall not be compromised by the presence of the anticipated worst case natural electron environment. To allow for the fact that the trapped particle population is not isotropically distributed, a 2% peak-to-peak sinusoidal variation with angle shall be assumed superimposed on the average electron environment. The representation for the worst case natural environment assumes isotropic flux with the spectral distribution shown in Table 9.1.2.4.

Table 9.1.2.4
Assumed Worst-Case Electron Environment

E (MeV)	0.3	0.45	1.05	1.9
J(>E) (p cm^{-2} sec^{-1})	2×10^7	7×10^6	7×10^5	1.5×10^5

9.1.2.5 ***Temporal Resolution*** - The temporal resolution of the X-ray fluxes shall be at least 3 seconds and the flux values shall be transmitted to ground in real-time. The time delay between flux measurements in the two X-ray channels shall be less than 0.1 second. The instrument response to an instantaneous change in X-ray flux (a step function) shall be such that the telemetered output of each channel shall be within 10% of its final value within the specified temporal resolution. The analytical channel response function shall be specified by the spacecraft contractor to sufficient accuracy to allow for possible ground correction of the data. Each EUV channel shall be sampled at least every 30 seconds.

9.1.2.6 ***Angular Response*** - The instrument response, including the combined effect of spacecraft pointing and sensor field-of-view (FOV), shall not deviate by more than 5%, with a goal of 2%, for point sources of constant flux within 20 arc-minutes of the solar disk center.

9.1.2.7 ***Pointing Knowledge*** - The spacecraft contractor shall provide a method of determining the XRS/EUV mechanical pointing with respect to the Sun-center. This pointing information shall be provided at the same cadence as the measurements made by the XRS/EUV, and shall be sufficient to determine unambiguously the direction to the Sun whenever the Sun center is within ±10° of the mechanical axis of the XRS/EUV FOV (i.e., multiple viewing angles should not give the same output signal). The accuracy of the XRS/EUV pointing determination with respect to the Sun-center shall be ±2 arcminutes with a resolution of ±1 arcmin when the Sun-center is within ±2° of the mechanical axis of the XRS/EUV FOV. This high accuracy and resolution is not required at viewing angles > 2°. The spacecraft contractor shall provide a calibration of the XRS/EUV FOV with respect to the mechanical axis used for the pointing determination.

9.1.2.8 ***In-flight Calibration*** **-** A calibration mode shall be provided by ground command for determining electronic processing gain to an accuracy of 2% for all channels and for verifying basic instrument operation. This calibration shall be traceable to incident flux through the detector pre-flight calibration. The calibration shall be both self-terminating and able to be terminated by ground command.

9.1.2.9 ***Pre-flight Calibration*** **-** Calibration of the instrument response to flux shall cover the full dynamic range and expected range of sensor temperatures. Trend sensitivity data shall be provided throughout the instrument and spacecraft level tests.

9.1.2.9.1 ***Wavelength Response*** **-** The wavelength response of each channel shall be specified to an accuracy of ±5%. The instrument shall be calibrated at sufficient wavelengths to adequately characterize the wavelength response from each detector component with a wavelength-dependent response. An analysis of the uncertainties and errors of each component shall be provided to verify that the number of calibration wavelengths is sufficient to adequately characterize the relative wavelength response to an accuracy of ±5%.

9.1.2.9.2 ***Absolute Response*** **-** The accuracy of the telemetered X-ray flux data shall be demonstrated by analysis and/or test to be better than ±20% with a goal of ±10% of the actual flux for flux values greater than 20 times threshold. The accuracy of the telemetered EUV flux data shall be better than <u>±20% with a goal</u> +/- 10% of the actual flux. CCR6057A Mod32

9.1.3 ***Energetic Particle Sensor (EPS)***

9.1.3.1 ***Solar Protons*** **-** The EPS instrument shall provide proton flux measurements in the energy range from 0.8 MeV to greater than 500 MeV. The flux shall be determined in at least seven logarithmically spaced energy intervals within this range. The instrument shall have a minimum of two look-directions, each with a FOV containing at least ±30° in azimuth (within the equatorial plane) and at least ±30° in elevation (30° above and 30° below the equatorial plane). The FOVs shall be oriented such that gaps in azimuthal coverage are no larger than 120°. If two look-directions are used, it is desirable that one be oriented eastward and one westward.

9.1.3.2 ***Alpha Particles*** **-** The instrument shall provide alpha particle flux measurements in the energy range from 3.8 to 400 MeV. The flux shall be determined in at least six logarithmically spaced energy intervals within this range.

9.1.3.3 ***Maximum Proton and Alpha Flux to be Measured*** **-** The detector shall resolve the largest likely solar particle event. The spectrum for this event can be represented as:

$$J(>E) = 10^6 E^{-1} \text{ protons/(cm}^2\text{ sec sr)} \qquad (0.8 \text{ MeV} < E < 10 \text{ MeV})$$

$$J(>E) = 10^7 E^{-2} \text{ protons/(cm}^2\text{ sec sr)} \qquad (E > 10 \text{ MeV})$$

with a proton-to-alpha number flux ratio greater than four. The output shall not decrease as the flux increases up to three times this maximum flux.

9.1.3.4 ***Minimum Proton and Alpha Flux to be Resolved*** **-** The detector shall resolve a flux of approximately

$J = 0.7$ particles / (cm^2 sec sr) (0.8 MeV < E < 10 MeV)

$J = 4.0\ E^{!0.8}$ particles / (cm^2sec sr) (E > 10 MeV)

for both protons particles and alpha particles. This minimum flux must be resolved with a minimum of 10 counts above background (30% statistical uncertainty) in each energy channel over a 5-minute interval.

9.1.3.5 ***Magnetospheric Protons*** **-** The detector shall resolve the largest likely proton flux in the energy range from 80 keV to 800 keV. The flux shall be determined in at least five logarithmically spaced differential energy channels.

The instrument shall resolve the angular distribution of proton directional flux over these energies. At least five azimuthal bins (in the equatorial plane) spanning a total FOV of at least 170° and five elevation bins spanning an FOV of at least 170° (centered on the equatorial plane) shall be sampled. This resolution could be accomplished with a total of nine angle bins; five in azimuth and five in elevation with the center bin common to both planes of coverage.

9.1.3.6 ***Maximum Magnetospheric Proton Flux to be Measured*** **-** The detector shall resolve the largest likely proton flux in the energy range from 80 keV to 800 keV. The spectrum for this flux level can be represented as:

$J(>E) = 400.\ E^{-3.5}$ protons/(cm^2 sr sec) (E in MeV)

The output shall not decrease as the flux increases up to three times this maximum flux.

9.1.3.7 ***Minimum Magnetospheric Proton Flux to be Resolved*** **-** The detector shall resolve flux levels given by:

$J(>E) = 0.3\ E^{-2.4}$ protons/(cm^2 sr sec) (E in MeV)

This minimum flux must be resolved with a minimum of 10 counts rounded to the nearest integer above background (30% statistical uncertainty) in each energy channel over a 5-minute interval.

9.1.3.8 ***Electrons*** **-** The instrument shall provide electron flux measurements in the energy range from 30 keV to greater than 4 MeV. The electron flux at energies from 30 keV to 600 keV shall be determined in at least five logarithmically spaced differential channels. For electron energies above 600 keV, there shall be at least three integral electron channels with threshold energies of 0.6 MeV, 2.0 MeV, and 4.0 MeV. The threshold energy is that energy at which the detection efficiency first exceeds from 10 to 30% of the peak detection efficiency in the channel. The energy ranges of the integral channels have been chosen to provide continuity with prior measurements.

For electron energies greater than 0.6 MeV, the EPS shall have a minimum of two look-directions, each with a FOV spanning at least ±30° in azimuth (within the equatorial plane) and at least ±30° in elevation (30° above and 30° below the equatorial plane). The FOVs shall be oriented such that gaps in azimuthal coverage are no larger than 120°. If two look-directions are used, it is desirable that one be oriented eastward and one westward.

For electron energies less than 0.6 MeV, the instrument shall resolve the angular distribution of electron directional flux in at least five azimuthal bins (in the equatorial plane) spanning a total FOV of at least 170° and five elevation bins spanning at least a 170° FOV (centered on the equatorial plane).

9.1.3.9 ***Maximum Electron Flux to be Measured*** - The detector shall resolve the largest likely electron flux. The spectrum for this flux level can be represented as:

$J(>E) = 5x10^5 E^{-1.8}$ electrons/(cm^2 sr sec) (E less than 2 MeV)

$J(>E) = 7x10^5 E^{-2.3}$ electrons/(cm^2 sr sec) (E greater than 2 MeV)

The output shall not decrease as the flux increases up to three times this maximum flux.

9.1.3.10 ***Minimum Electron Flux to be Resolved*** - The detector shall resolve flux levels given by:

$J(>E) = 4.5E^{-2.2}$ electrons/(cm^2 sr sec) (E in MeV)

to a minimum flux level of J = 1 electron/(cm^2 sec sr). This minimum flux must be resolved with a minimum of 10 counts above background (30% statistical uncertainty) in each energy channel over a 5-minute interval.

9.1.3.11 ***Sampling Rate*** - Each of the data channels shall be sampled at least once every 33 seconds.

9.1.3.12 ***Noise*** - Noise shall not widen the effective response by more than 10 keV for electrons and protons at the thresholds below 100 keV. Above 100 keV, noise shall not widen the effective response by more than 10% of the threshold energies.

9.1.3.13 ***Stability*** - The electronic thresholds defining the energy band edges (at the limits of the dynamic range) shall not change by more than 3% over the predicted operating conditions.

9.1.3.14 ***Resolution*** - Data compression shall accommodate the maximum accumulation count in one word of telemetry. A compression algorithm, having a maximum compression error of no worse than that resulting from pseudolog compression of 19 to 8 bits using 4 bits of mantissa and 4 bits of exponent, shall be used. For this specification, the compression error is defined as the difference between the accumulator count and the count reconstructed from the compressed output, divided by the accumulator count.

9.1.3.15 ***In-flight Calibration*** - A calibration mode shall be provided to verify basic instrument operation and to determine the value of energy band edges. Electronic discriminator threshold levels shall be determined to ±5%. The calibration shall be both self-terminating and able to be terminated by ground command.

9.1.3.16 ***Ground Calibration*** - Sensor components (detectors, modulators, collimators, etc.) shall be calibrated using a combination of accelerators and/or nuclear sources to determine actual incident particle energy thresholds, absorbing properties, and/or geometric properties. Solid-state detector calibration shall, as a minimum, determine the nuclear thickness and dead layer.

The energy-dependent and the directional responses of the sensors shall be determined for energies ranging from the detector's low-energy threshold to energies for which the particle flux is below the instrument detection threshold, assuming particle flux levels given by the maximum particle fluxes specified in sections 9.1.3.3, 9.1.3.6, and 9.1.3.9.

9.1.3.17 ***Contaminants*** - The response of the various data channels to particles out-of-aperture or of a different species or energy shall be minimized and shall be determined by analysis and/or test. The spacecraft contractor shall provide correction algorithms for the out-of-aperture and out-of-band response of all channels to the full range of flux levels specified in Sections 9.1.3.3, 9.1.3.4, 9.1.3.6, 9.1.3.7, 9.1.3.9, and 9.1.3.10.

9.1.3.18 ***Radiation Damage*** - The spacecraft contractor shall demonstrate the steps taken to minimize and/or mitigate the effects of radiation damage on the sensor systems during the design lifetime of the spacecraft.

9.1.4 ***High Energy Proton and Alpha Detector (HEPAD)*** - Measurements of the proton flux above 350 MeV and of the alpha-particle flux above 640 MeV/nucleon shall be provided.

9.1.4.1 ***Spectral Bands*** - Proton flux shall be measured in at least three contiguous, differential energy bands, between approximately 350 MeV and 700 MeV or above, together with an integral band above the upper differential band. Alpha particle flux shall be measured in at least one band from approximately 640 MeV/nucleon to 850 MeV/nucleon and one integral band above the differential band limit.

9.1.4.2 ***Field of View*** - The detector acceptance aperture shall have a half angle greater than 24°. The aperture shall be centered within 5° of the equatorial plane and within 100° of the local zenith (radially outward from Earth).

9.1.4.3 ***Geometric Factor*** - The geometric factor shall be no less than 0.9 (cm^2-sr).

9.1.4.4 ***Singles Channels*** - For a design similar to the current HEPAD instrument, primary data channels shall be supported by additional channels as necessary. As a minimum, there shall be a single channel at the lowest threshold of each separate detector used and a fast coincidence channel from the lowest threshold of each separate detector.

9.1.4.5 ***Accumulation Efficiency*** - The accumulation efficiency is defined as the percentage of events detected that are actually accumulated and further processed. The accumulation efficiency for the two highest energy proton channels shall not be less than 80%. The accumulation efficiency for the remaining four primary data channels shall not be less than 40%. Accumulation efficiency of the singles channels shall be adequate for their use in determining reliability of the data from the primary proton and alpha particle channels.

9.1.4.6 ***Stability and Accuracy*** - The data channel energy thresholds and the intensity measurement shall be accurate to better than 15% over the expected operating temperature range and supply voltage range.

9.1.4.7 ***Data Rate*** - Each of the six primary data channels shall be measured at least once every 60 seconds. Each of the singles channels shall be measured at least once every 300 seconds. Accumulation intervals for the singles channels shall be coincident with the primary data channels.

9.1.4.8 ***Count Resolution*** - Data compression shall be used to accommodate the maximum accumulator count in one word of telemetry. A compression algorithm having a maximum compression error no worse than that resulting from pseudolog compression of 19 to 8 bits using 4 bits of mantissa and 4 bits of exponent shall be used. The compression error for this specification is defined as the difference between the accumulator count and the count reconstructed from the compressed output, divided by accumulator count.

9.1.4.9 ***In-flight Calibration*** **-** An in-flight calibration (IFC) system or method shall be provided to verify basic instrument operation, the accuracy of data channel energy thresholds, and intensity measurements. The calibration shall be both self-terminating and able to be terminated by ground command.

9.1.4.10 ***Contaminants*** **-** The primary data channel responses to penetrating electron radiation shall be calibrated separately for the energy range of 2 to 13 MeV/electron. The instrument's maximum electron flux environment is defined by the integral energy spectrum:

$$J(>E) = 7\times10^{5} E^{-2.3} \text{ electrons/(cm}^2 \text{ sr sec)} \qquad (2 \text{ MeV} < E < 13 \text{ MeV}).$$

Proton contamination in the alpha channels shall be < 0.1% for the total passband. Compliance with this specification shall be demonstrated both during ground test in a suitable facility and during the performance of muon spectra runs.

9.1.4.11 ***Lifetime*** **-** Ground commands may be used to compensate for degradation mechanisms in the flight hardware if they are needed to meet spacecraft lifetime requirements.

9.1.4.12 ***Maximum Flux to be Measured*** **-** The instrument shall meet all specifications in measuring a spectrum given by:

$$J(>E) = 10^{7} E^{-2} \text{ protons/(cm}^2 \text{ sec sr)} \qquad (E \text{ in MeV})$$

with a proton-to-alpha ratio of four.

9.2 Imager Interface

The spacecraft shall accommodate either an Imager or an Advanced Imager instrument. In either case, the spacecraft shall comply with the following interface requirements in addition to the requirements in the Imager ICD:.

9.2.1 ***General Requirements*** **-** The spacecraft shall meet all spacecraft-instrument interface requirements defined in the Imager ICD and applicable Imager interface control drawings. The spacecraft shall allow Imager operations independent of all other instruments, and the Imager shall be capable of operating and transmitting data during eclipses.

9.2.2 ***Mechanical Interface*** **-** The spacecraft shall meet the following additional mechanical interface requirements:

9.2.2.1 ***Envelope & Mass*** **-** The spacecraft and launch vehicle design shall accommodate an Imager sensor module, electronics module, and power supply of the mass and dimensions defined in the Imager ICD. The spacecraft shall include margin for a growth in the total mass of the Imager up to 310.6 lb (140.9 Kg). (Deleted) CCR4166 Mod 32

9.2.2.2 ***Access Provisions*** - The spacecraft-Imager interface design shall allow the removal of any individual Imager module without removal of other instrument modules or spacecraft components. The spacecraft-Imager interface design shall also allow removal of spacecraft modules and components without removing or disturbing any Imager module.

9.2.3 ***Electrical Interface*** **-** The spacecraft shall meet the following additional electrical interface requirements:

9.2.3.1 ***Cross-strapping*** - All inputs and outputs between the Imager and the spacecraft shall be redundant, including command, mirror compensation, and data and telemetry transmission. The spacecraft shall provide cross-strapping to operate side one or two of the Imager electronics with either side of the spacecraft electronics.

9.2.3.2 ***Power Consumption*** **-** The spacecraft shall provide power for Imager operations as defined in the Imager ICD. The spacecraft shall provide 6% power margin over the sum of the Imager and Sounder average on-orbit power consumption while scanning a frame as defined in 3.4.2.3.3 Average Power Draw While Scanning section of the Imager and Sounder ICDs during initial outgas and normal operation modes at BOL and EOL.

The power requirements stated in the Imager ICD for thermal control are dependent on the spacecraft to instrument interface design and the attitude of the instruments relative to the Sun during launch, transfer orbit, and on-orbit storage. The requirements are considered satisfied if the power supplied is sufficient to maintain the instrument components within their mission allowable temperatures.

9.2.3.3 ***Static Charging*** - Precautions shall be employed to ensure electrostatic discharge (ESD) is avoided and static charging on surfaces adjacent to or near the instrument are minimized through selection of materials and implementation of appropriate processes.

9.2.3.4 ***Electromagnetic Interference (EMI)*** - The spacecraft shall comply with the EMI requirements specified in the Imager ICD. In the case of a conflict between the ICD and Section 8.4 of this specification, the ICD shall govern.

9.2.4 ***Command Interface*** **-** The spacecraft shall process and distribute command signals to the Imager. The spacecraft-Imager command interface shall be consistent with the data content and signal characteristics defined by the Imager ICD. The spacecraft shall include margin for a 10% growth in the number of Imager pulse commands.

9.2.5 ***Data Handling***

9.2.5.1 ***Wideband Data Transmission*** **-** The spacecraft shall transmit the Imager wideband data stream continuously via the sensor data downlink. Data shall be transmitted to the ground station within 29 seconds of its receipt from the Imager. The spacecraft-Imager wideband interface shall be consistent with the data content and signal characteristics defined by the Imager ICD. The spacecraft shall include margin for a 12.5% increase in the wideband data transmission rate and the GVAR data relay rate.

9.2.5.2 ***Health, Safety, and Status Telemetry*** **-** The spacecraft shall sample and transmit the Imager analog and bi-level telemetry defined in the Imager ICD. The spacecraft shall include margin for a 5 % growth in the number of telemetry points from the Imager.

9.2.5.3 ***Servo Data Transmission*** **-** The spacecraft shall sample the Imager analog east-west and north-south servo error telemetry (defined in the Imager ICD) and telemeter these data via the MDL.

9.2.6 ***Image Navigation and Registration (INR) System Interface*** **-** The spacecraft shall meet all INR interface requirements as defined by the Imager ICD.

9.2.6.1 ***Imager Thermal Distortion*** **-** The spacecraft-Imager interface shall be designed to limit the diurnal structural distortion of the Imager line of sight relative to the primary spacecraft attitude reference to be compatible with the overall INR performance requirements and the dynamic range of the Imager Image Motion Compensation (IMC) signal.

9.2.6.2 ***INR Electrical Interface***

9.2.6.2.1 ***Scan Compensation to Imager*** **-** The spacecraft may provide east-west and north-south scan compensation signals to correct for Imager line of sight attitude deviations to meet the system-level INR requirements. If these signal paths are employed by the spacecraft INR design, the scan compensation signal shall comply with the interface requirements defined by the Imager ICD.

9.2.6.2.2 ***INR Data to Imager*** **-** The Imager can accept digital data from the INR subsystem, package the data in a header or trailer data block, and downlink the data within the wideband data stream. If this signal path is employed by the INR design, it shall comply with the interface requirements defined by the Imager ICD and the applicable Imager interface control, harness, and wire list drawings.

9.2.6.2.3 ***Imager INR Data to Spacecraft*** **-** The Imager can provide scan status information to the spacecraft. The definition of this data is provided in the Imager ICD. If these signal paths are employed by the spacecraft INR design, they shall comply with the interface requirements defined by the Imager ICD and the applicable Imager interface control, harness, and wire list drawings.

9.3 Sounder Interface

The spacecraft shall accommodate either a Sounder or an Advanced Sounder instrument. In either case, the spacecraft shall comply with the following interface requirements in addition to those in the Sounder ICD:

9.3.1 ***General Requirements*** **-** The spacecraft shall meet all spacecraft-instrument interface requirements defined the Sounder ICD and applicable Sounder interface control drawings. The spacecraft shall allow Sounder operations independent of all other instruments, and the Sounder shall be capable of operating and transmitting data during eclipses.

9.3.2 ***Mechanical Interface -*** The spacecraft shall meet the following additional mechanical interface requirements:

9.3.2.1 ***Envelope and Mass*** - The spacecraft and launch vehicle design shall accommodate a Sounder sensor module, electronics module, and power supply of the mass and dimensions defined in the Sounder ICD. The spacecraft shall include margin for a growth in the total mass of the Sounder up to 151.0 kg (333.0 lb). (Deleted) CCR 4139 Mod 24 Mod 28-Adm CCR4166 Mod 32.

9.3.2.2 ***Access Provisions*** - The spacecraft-Sounder interface design shall allow the removal of any individual Sounder module without removal of other instrument modules or spacecraft components. The spacecraft-Sounder interface design shall also allow removal of spacecraft modules and components without removing or disturbing any Sounder module.

9.3.3 ***Electrical Interface*** - The spacecraft shall meet the following additional electrical interface requirements:

9.3.3.1 ***Cross-strapping*** - All inputs and outputs between the Sounder and spacecraft shall be redundant, including command, mirror compensation, and data and telemetry transmission. The spacecraft shall provide cross-strapping to operate side one or two of the Sounder electronics with either side of the spacecraft electronics.

9.3.3.2. ***Power Consumption*** - The spacecraft shall provide power for Sounder operations as defined in the Sounder ICD. The spacecraft shall provide 6% power margin over the sum of the Imager and Sounder average on-orbit power consumption while scanning a frame as defined in 3.4.2.3.3 Average Power Draw While Scanning section of the Imager and Sounder ICDs during initial outgas and normal operation modes at BOL and EOL.

The thermal control power requirements stated in the Sounder ICD are dependent on the spacecraft-to-instrument interface design and the attitude of the instruments relative to the Sun during launch, transfer orbit, and on-orbit storage. The requirements are considered satisfied if the power supplied is sufficient to maintain the instrument components within their MATs.

9.3.3.3 ***Static Charging*** - Precautions shall be employed to ensure ESD is avoided and static charging on surfaces adjacent to or near the instrument are minimized through selection of materials and implementation of appropriate processes.

9.3.3.4 ***Electromagnetic Interference (EMI)*** - The spacecraft shall comply with the EMI requirements specified in the Sounder ICD. In the case of a conflict between the ICD and Section 8.4 of this specification, the ICD shall govern.

9.3.4 ***Command Interface*** - The spacecraft shall process and distribute command signals to the Sounder. The spacecraft-Sounder command interface shall be consistent with the data content and signal characteristics defined by the Sounder ICD. The spacecraft shall include margin for a 10% growth in the number of Sounder pulse commands.

9.3.5 ***Data Handling***

9.3.5.1 ***Wideband Data Transmission*** - The spacecraft shall transmit the Sounder wideband data stream continuously via the sensor data downlink. Data shall be transmitted to the ground station within 29 seconds of its receipt from the Sounder. The spacecraft-Sounder wideband telemetry interface shall be consistent with the data content and signal characteristics defined by he Sounder ICD. The spacecraft shall include margin for an increase in the Sounder wideband data transmission rate of 1800% (18-fold), and a corresponding increase in the GVAR data relay rate.

9.3.5.2 ***Health, Safety, and Status Telemetry*** - The spacecraft shall sample and transmit the Sounder analog and bi-level telemetry defined in the Sounder ICD. The spacecraft shall include margin for a 5% growth in the number of telemetry points from the Sounder.

9.3.5.3 ***Servo Data Transmission*** **-** The spacecraft shall sample the Sounder analog east-west and north-south servo error telemetry (defined in the Sounder ICD) and telemeter these data via the MDL.

9.3.6 ***INR System Interface*** - The spacecraft shall meet all INR interface requirements as defined by the Sounder ICD.

9.3.6.1 ***Sounder Thermal Distortion*** - The spacecraft-Sounder interface shall be designed to limit the Sounder line of sight diurnal structural distortion relative to the primary spacecraft attitude reference to be compatible with the overall INR performance requirements and the dynamic range of the Sounder IMC signal.

9.3.6.2 ***INR Electrical Interface***

9.3.6.2.1 ***Scan Compensation to Sounder*** **-** The spacecraft may provide east-west and north-south scan compensation signals to correct for Sounder line of sight attitude deviations to meet the system-level INR requirements. If these signal paths are employed by the spacecraft INR design, the scan compensation signal shall comply with the interface requirements defined by the Sounder ICD.

9.3.6.2.2 ***INR Data to Sounder*** **-** The Sounder <u>shall</u> accept digital data from the INR subsystem, package the data in a data block, and transmit the block within the wideband data stream. The digital data format from the INR subsystem shall be contractor specified and agreed to by the government. The design, shall comply with the interface requirements defined by the Sounder ICD and the applicable Sounder interface control, harness, and wire list drawings.

9.3.6.2.3 ***Sounder INR Data to Spacecraft*** **-** The Sounder can provide scan status information to the spacecraft. The definition of this data is provided in the Sounder ICD. If these signal paths are employed by the spacecraft INR design, they shall comply with the interface requirements defined by the Sounder ICD and the applicable Sounder interface control, harness, and wire list drawings.

9.4 ***Solar X-Ray Imager Interface***

9.4.1 ***General Requirements*** - The spacecraft shall meet all spacecraft-instrument interface requirements defined by the Solar X-ray Imager (SXI) ICD S-415-25. The spacecraft shall allow SXI operations independent of all other instruments, and the SXI shall be capable of operating and transmitting data during eclipses. In addition to the SXI ICD requirements, the spacecraft shall meet the following interface requirements:

9.4.2 ***Mechanical Interface***

9.4.2.1 ***Envelope & Mass*** **-** The spacecraft and launch vehicle design shall accommodate an SXI telescope and three electronic boxes of the mass and dimensions defined in the SXI ICD. The spacecraft shall be capable of accommodating a 1.9 lb (0.86 Kg) margin in the total mass of the SXI telescope and three electronics boxes. The spacecraft shall be capable of accommodating a 5% margin in the moments of inertia and center of gravity of the SXI telescope. In addition, the spacecraft shall be capable of accommodating a 10% margin in any or all dimensions of the SXI telescope and the three electronics boxes.

9.4.2.2 ***Access Provisions*** - The spacecraft-SXI interface design shall allow the removal of the SXI telescope and boxes without the removal of other instruments or spacecraft components. The spacecraft-SXI interface design shall also allow removal of spacecraft components without removing or disturbing the SXI telescope and boxes.

9.4.3 ***Electrical Interface***

9.4.3.1 ***Cross-strapping*** - All inputs and outputs between the SXI and spacecraft shall be redundant, including power, command, and data and telemetry transmission. The spacecraft shall provide cross-strapping to operate the SXI electronics with either side of the spacecraft electronics.

9.4.3.2 ***Power Consumption*** - The spacecraft shall have the capability to provide power for SXI operations as defined in the SXI ICD, plus the following margins:

1. Non-operational modes (launch, orbit raising, initial outgas, and storage): 20% margin over the non-operational power requirement defined in the GOES N-Q Spacecraft - SXI ICD.
2. Normal operations, non-eclipse: 20 Watts.
3. Normal operations, eclipse: 20 Watts.

9.4.3.3 ***Static Charging*** - Precautions shall be employed to ensure ESD is avoided and static charging on surfaces adjacent to or near the instrument are minimized through selection of materials and implementation of appropriate processes.

9.4.3.4 ***Electromagnetic Interference (EMI)*** - The spacecraft shall comply with the EMI requirements specified in the SXI ICD.

9.4.4 ***Command Interface*** - The spacecraft shall process and distribute command signals to the SXI. The spacecraft-SXI command interface shall be consistent with the data content and signal characteristics defined by the SXI ICD.

9.4.5 ***Data Handling***

9.4.5.1 ***Data Transmission*** - The spacecraft shall transmit the SXI data stream continuously via the MDL. Data shall be transmitted to the ground station at a rate of 100 kbps.

9.4.5.2 ***Health, Safety, and Status Telemetry*** - The spacecraft shall sample and transmit the SXI telemetry defined in the SXI ICD. The spacecraft shall be capable of accommodating a $\geq$5% margin in the number of telemetry points from the SXI.

9.4.6 ***Sun Pointing Capability*** - The spacecraft shall meet the pointing requirements of section 9.4.6 and all of its subsections, except during stationkeeping, eclipses, station changes greater than 1° per day, and housekeeping. The spacecraft shall provide for all instrumentation needed to demonstrate the pointing accuracy and pointing stability requirements are met per CDRL SDA-3.2.14-03. When E-W or N-S pointing control is performed, the control shall be achieved in a manner that is of negligible consequence to the pointing of the spacecraft or other instruments on the spacecraft.

The E-W and N-S pointing requirements are provided to maintain observations of the solar disk and off-limb phenomena within the SXI's FOV. The alignment of the SXI axes to the spacecraft are required to eliminate image smear induced by misalignment of the SXI detector's IMC and the solar image drift caused by the spacecraft motion with respect to the Sun. Note that this alignment does not refer to the Sun's axes, which are not co-aligned to the Earth's rotation axis. The solar coordinates will be determined from knowledge of the spacecraft orientation with respect to the Earth's rotation axis. The solar coordinates will be derived from the known relationships between coordinate systems of the Sun and Earth (e.g., the Naval Observatory's *The Astronomical Almanac* or J. Meeus's *Astronomical Formulae for Calculators,* Willman-Bell, Inc.).

9.4.6.1 ***East-West and North-South Pointing*** - See SXI ICD.

9.4.6.2 ***East-West and North-South Bias Pointing*** - See SXI ICD.

9.4.6.3 ***East-West and North-South Pointing Knowledge*** - See SXI ICD.

9.4.6.4 ***Pointing Stability*** - See SXI ICD.

9.4.6.5 ***Image Synchronization*** - See SXI ICD.

9.4.6.6 ***Rotation of the Detector Focal Plane*** - See SXI ICD.

9.4.6.6.1 ***SXI Y- and Z-axis Alignment*** - See SXI ICD.

9.4.6.6.2 ***SXI Rotation about the X-axis*** - See SXI ICD.

9.5 Instrument of Opportunity (IO) Interface

9.5.1 ***General Requirements*** **-** The spacecraft shall allow IO operations independent of all other instruments, and the IO shall be capable of operating and transmitting data during eclipses. The spacecraft shall meet all IO to spacecraft interface requirements as defined by the IO ICD. The spacecraft-IO interface shall be developed jointly by the instrument and spacecraft contractors.

9.5.2 ***Mechanical Interface***

9.5.2.1 ***Envelope*** **-** The spacecraft design shall accommodate the IO optical bench and spacecraft bus mounted modules. The dimensions of the optical bench-mounted module are 50 x 37.5 x 70 cm., the longest dimension inclusive of a sunshade or optical baffle. The dimensions of a spacecraft bus-mounted module are 37.5 x 37.5 x 37.5 cm.

9.5.2.1.2 ***Mass*** **-** The mass of the optical bench mounted module shall be no more than 25.0 kg. The total mass of the IO (optical bench and spacecraft mounted modules) shall be 35.0 kg or less. The mass does not include the harnesses, mechanical hardware, or thermal control materials used at the mounting interface.

9.5.2.2 ***Field of View*** **-** The spacecraft-LM interface shall be designed to provide the LM an unobstructed FOV within ±10° of optical nadir.

9.5.2.3 ***Structural Mounting Characteristics***

9.5.2.3.1 ***Mounting*** - The LM electronics module shall be mounted internal to a Faraday cage provided by the spacecraft.

9.5.2.3.2 ***Design Load Factors*** - The quasi-static design load factors for the LM sensor and electronics modules are 15 g's in all three axes. The spacecraft contractor may assume the center of gravity is at the center of mass for both modules.

9.5.2.3.3 ***Alignment*** - The LM sensor module optical axis shall be aligned to the center of the Earth disk to within ±0.1°.

9.5.2.4 ***Access Provisions*** - The spacecraft-LM interface shall allow the removal of any individual LM module without the removal of the other instrument modules or spacecraft components. The spacecraft-LM interface design shall allow removal of spacecraft modules without removing or disturbing any LM module.

9.5.3 ***Thermal Interface*** - The spacecraft mounting panel in contact with the LM sensor and electronics module shall be maintained within the MAT range of -20C to +20C while the module is dissipating up to 120 watts over the spacecraft life and for all orbital conditions. The spacecraft mounting panel in contact with the LM power supply module shall be maintained within the MAT temperature range of -10C to +40C while the module is dissipating up to 30 watts. Maximum operational dissipation shall not exceed 150 watts.

9.5.4 ***Electrical Interface***

9.5.4.1 ***Physical Electrical Interface***

9.5.4.1.1 ***Cross-strapping*** - All inputs and outputs between the LM and spacecraft shall be redundant, including power, command, and telemetry.

9.5.4.1.2 ***Test Point Access*** - The spacecraft-LM interface shall be designed to permit access to the LM test point connectors during spacecraft-level testing.

9.5.4.2 ***Power Consumption*** - The spacecraft shall provide an average power of 150 watts to the LM during normal on-orbit operations. The spacecraft shall provide an average power of 15 watts to the LM during transfer orbit and on-orbit storage mode.

9.5.4.3 ***Time Code*** - The spacecraft shall provide the LM a time code which can be correlated to within one millisecond of Universal Time (UT). The spacecraft shall distribute time information, including any factors necessary to correlate spacecraft time to UT, to the LM at a 1 Hz rate. The exact time tagging will be developed mutually by the spacecraft and LM contractors.

9.5.5 ***Command Interface*** - The spacecraft shall process and distribute commands to the <u>IOO</u>. <u>One serial and six pulse command channels shall be provided to each IOO (one or two instruments).</u> CCR6043 Mod 36

9.5.6 ***Data Handling***

9.5.6.1 ***Science Data*** - The spacecraft shall transmit the 100 kbps, NRZ-L formatted IOO data stream continuously to the ground via the MDL. If two instruments of opportunity are installed, the aggregate data rate from both instrument shall not exceed 100 kbps. Additionally, the data rate from IOO #2 shall not exceed 50 kbps. CCR6043 Mod 36

9.5.6.1.1 ***Science Data Electrical Interface*** - The spacecraft shall be capable of processing up to two (2) separate science data output streams from the instrument of Opportunity set, at the combined data rate (deleted part of sentence) specified in section 9.5.6.1. Each of the two data streams shall consist of two separate data and clock signals and these signals shall have the following electrical characteristics: CCR6043 Mod 36

1. Signal Format: NRZ-L
2. Signal Type: EIA-RS-422

9.5.6.2 ***Telemetry Data*** - The spacecraft shall sample IOO telemetry and include the data in the spacecraft CCR6043 Mod 36 telemetry (PCM) downlink. The spacecraft shall provide one 8-bit analog and nine 8-bit passive (conditioned) analog channels, each sampled at a minimum rate of 0.05 Hz, in the spacecraft telemetry (PCM) format.

9.5.7 ***Pointing Requirements***

9.5.7.1 ***Pointing Knowledge*** - The knowledge of the LM pointing attitude at the sensor module interface to the spacecraft during the normal operational mode shall be accurate to within ±112 Fradians, 3ó. This tolerance represents the root-sum-square total of the pointing knowledge uncertainties due to all effects, including spacecraft thermal deformations, alignment uncertainties, etc. The spacecraft contractor can assume that a one-time post-launch measurement of the LM pointing can be used to calibrate launch-induced changes in pointing.

9.5.7.2 ***Pointing Stability*** - The maximum excursion, peak-to-peak, of the LM sensor module interface during the normal operational mode shall be ±35 Fradians, 3ó, over a time interval of one second.

10.0 SPACECRAFT SUBSYSTEMS

Section 10.1 addresses the GOES N-Q spacecraft telemetry and command (T&C) system, and section 10.2 addresses requirements of the communications subsystem. Attitude control, propulsion, power and electrical, thermal control, structures, mechanisms, flight software, on-board computer, and contamination are presented in sections 10.3 through 10.11, respectively.

The T&C subsystem comprises the spacecraft signal paths used for commanding the spacecraft, telemetering spacecraft health and safety and SEM instrument data, and performing spacecraft ranging. The communication subsystem comprises all signal paths used for relaying and telemetering environmental data: raw Imager and Sounder; MDL (SXI and other data such as instrument servo error and ADS data); processed Imager and Sounder (GVAR) data, WEFAX and EMWIN broadcasts; SAR; and DCPI and DCPR. Table 10.0 provides the requirements of the radio frequency (RF) link parameters of the GOES N-Q T&C and communication systems.

10.1 Telemetry and Command (T&C)

Section 10.1.1 states the general requirements of the T&C system for on-orbit operations only, divided into telemetry, command, and ranging functions. These requirements are then amplified in the command and telemetry baseband and RF sections. The spacecraft contractor shall define the LOR T&C requirements.

10.1.1 ***General Requirements* -** The T&C function shall provide the capability to telemeter spacecraft and instrument health and safety and SEM instrument data, telemeter the contents of on-board memories, uplink commands to the spacecraft and instruments in clear and encrypted mode, and perform DSN ranging. The T&C function shall be fully redundant, with no single points of failure. The subsystem shall perform DSN-compatible command, telemetry and ranging functions, CDA (1694 MHz) telemetry transmission, and NSA-approved command decryption. The GOES N-Q spacecraft shall provide the capability to transmit DSN and CDA telemetry streams simultaneously.

The command function shall be capable of processing real-time commands while concurrently executing long-duration commands (e.g., a solar array slew), and on-board computer uploads.

The telemetry function shall be capable of downlinking all health and safety and SEM instrument telemetry data in one stream (i.e., no commanding required to obtain all health and safety data), referred to herein as the normal stream. A capability for user-definable telemetry contents shall be provided. The transmission of a user-defined stream shall not preclude transmission of the normal stream. Data channels (telemetry parameters or points) shall be provided with the required accuracies and update rates to permit monitoring and trending of spacecraft and instrument health and safety and collection of SEM instrument data. It shall also be possible to command the telemetering of the contents of selected on-board memories and storage locations.

The ranging function shall coherently receive, demodulate, remodulate, and transmit a ground station ranging signal structure while preserving the signal integrity in the presence of noise. The ranging function shall operate simultaneously with the telemetry and command functions without degradation or interference.

Table 10.0

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GOES N-Q On-Orbit RF System Link Parameter Requirements

Link	DSN & CDA Telemetry & Command DSN Station	CDA Telemetry & Command CDAS	Sensor Data Raw Imager & Sounder SD	Processed Data Relay PDR / GVAR	Weather Facsimile WEFAX	Emergency Managers Info Network EMWIN	Multi-use Data Link MDL (SXI, ADS Data)	DCPI Platform Interrogate	DCPR Platform Report	Search & Rescue
Spec §	10.1	10.1	10.2	10.2	10.2	10.2	10.2	10.2	10.2	10.2
Satellite Tx										
EIRP (dBm)	Contractor	Contractor Spec	Contractor Spec	Contractor Spec	Contractor Spec	Contractor Spec	Contractor Spec	Contractor Spec	46	45.0
Link Co	Omni	Omni	Earth Coverage	Earth Coverage	EarthCoverage	Earth Coverage	Earth Coverage	Earth Coverage	Earth Coverage	Earth Coverage
Freq (MHZ)	2209.086	1694.000	1676.000	1685.700	1691.000	Contractor Spec.	1681.478	468.8125 Spare 468.8250 East 468.8375 West	1694.500 Domestic 1694.800 Int'l	1544.5
Polarization	RHC	RHC	Linear N-S	Linear N-S	Linear N-S	Linear N-S	Linear N-S	RHC	Linear N-S	RHC
Data Rate	1, 4 kbps	1,4 kbps	2.62 Mbps Im	2.11 Mbps	293 ksps Coded	25 kbps un-coded	Contractor Spec	100 bps	100 to 1200 bps	400 bps
Data Format	Bi-phase-L	Bi-phase-L	40 kbps Sdr						1.5kHz/Ch for 100 bps BPSK and 300 bps 8PSK 3.0kHz/Ch for 1.2 kbps 8PSK	
Sub carrier (Mhz) Carrier Modulation	1.024 BPSK Phase Mod	None PSK	QPSK 4:1	BPSK ±3°	BPSK ±3°	BPSK ±3°	QPSK	±60° BPSK	±60° BPSK for 100 bps BPSK Trellis codulated for 8 PSK	PM
Satellite Rx										
Link Cov'ge	Omni	Omni	N/A	Earth Coverage	Earth Coverage	Earth Coverage	N/A	Earth Coverage	Earth Coverage	Earth Coverage
Dyn Range (dBmi)	-115 to -50	N/A	N/A	-96 to -86	-107 to -97	-114 to -104	N/A	-114 to -104	Below Noise to -100	Below Noise to -125
Min Rx G/T (dB/K)	Contractor	Contractor Spec	N/A	Contractor Spec	Contractor Spec	Contractor Spec	N/A	Contractor Spec	-18.7	Contractor
Polarization	RHC	RHC	N/A	Linear N-S	Linear N-S	Linear N-S	N/A	Linear N-S	RHC	RHC
Freq (MHz)	2034.200	2034.200	N/A	2027.700	2033.000	Contractor Spec	N/A	2034.8875 Spare 2034.9000 East 2034.9125 West	401.900 402.200	406.050 406.025
Subcarrier (kHz)	16	16								
Data Rate	2 kbps/NRZ-	2 kbps/NRZ-L	N/A	2.11Mbps NRZ-S	293 ksps NRZ-	25 kbps NRZ-L	N/A	100 bps	100 to 1200 bps	400 bps
Modulation	BPSK/Phase Mod	BPSK/Phase Mod	N/A	BPSK +/- 3 deg. degrees	BPSK +/- 3 degs	BPSK +/- 3 degs.	N/A	±60° BPSK	±60° BPSK for 100 bps BPSK Trellis codulated for 8PSK	PM
Ground Tx										
Nom EIRP(dBm) Storage/safehold	101.2 from 114	88.7 -91.7 CDA 104.600	N/A	95	83.7	76.7	N/A	76.7	45.0	36.0
Polarization	RHC DSN	Linear N-S	N/A	Linear N-S	Linear N-S	Linear N-S	N/A	Linear N-S	RHC	Linear N-S
Ground Rx										
G/T (dB/K)	32.0 @ DSN 26.0 @ CDA	26.0 @ CDA	26.0 @ CDA	15.2 @ Users	-0.30 @ Users	-0.30 @ Users	15.2 @ SOCC 15.2 @ SEC	-21.6 @ Land -29 @ Buoys	12.0 @ Users	6.0 @ LUT
Polarization	RHC/DSN, Lin	Linear N-S	Linear N-S	Linear N-S	Linear N-S	Linear N-S	Linear N-S	RHC	Linear N-S	RHC
Rx System Loss (dB)	Contractor Spec	Contractor Spec	2.3	2.3	2.3	2.3	2.3	2.0	2.0 for 100 bps BPSK Not applicable for 8PSK	2.0
E-to-E BER	1E-05	1E-05	1E-08	1E-06	1E-08	1E-08	1E-08	1E-05	1E-06 for 100 bps BPSK Not applicable for 8PSK	1E-05

10.1.1.1 ***Redundancy*** **-** The telemetry, command, and ranging functions shall each consist of redundant channels. Each channel shall consist of an RF section and a baseband section. Each RF section shall consist of a command receiver, a DSN telemetry transmitter, and a CDA telemetry transmitter. Redundancy shall be maximized by cross-strapping the baseband sections with the RF sections. Loss of one section, any part of one section, or input bus power to a section shall not in any way affect or degrade the operation of the other sections.

10.1.1.2 ***Spare Capacity*** **-** The telemetry baseband section shall provide an expansion capability of at least 15% in the number of available data channels. The spare capacity shall consist of a mix of status, digital, and analog data channels. Spare command address space capacity of at least 25% shall be provided. These spare capacities shall be in addition to that specified for the GFE instruments and the baseline MDL instruments. The spare telemetry and command capacity shall be available at the time of the spacecraft preliminary design review (PDR).

10.1.1.3 ***Bus Under Voltage Dropout*** **-** The spacecraft contractor shall propose the maximum bus under voltage the T&C subsystem functions can experience without performance degradation.

10.1.2 ***Command Baseband Section*** **-** Each baseband section shall be identical and capable of receiving and executing a command on the spacecraft. Each baseband section shall have a unique command decoder address provided by the spacecraft contractor. No conceivable bus voltage or environmental condition shall result in spurious command reception or execution. Command data bit detection shall be inhibited in the presence of receiver noise and no carrier. No single, incorrect functioning of the command subsystem shall cause the failure of the spacecraft, or of a spacecraft component, or the loss of any spacecraft function.

10.1.2.1 ***Command Functions*** **-** Four different command modes shall be provided: real-time single , stored time-tagged single commands, real-time command sequences, and stored command sequences. All commands shall be verified by changes in telemetry data. Where extended duration, multiple execute command capability is required, the command subsystem shall remain capable of accepting, processing, and executing other subsystem commands. Any on-going command execution shall be capable of immediate termination. All command storage buffers shall be capable of ground initiated reset/clear. A command unit selection command shall clear all command buffers. There shall also be no command lockout.

10.1.2.1.1 ***Real-time Single Commands*** **-** Real-time single commands shall execute following verification of proper receipt by the command decoder. A spacecraft/ground mechanism shall prevent command execution out of sequence. A command protocol shall account (preferably via command count) for all real-time commands accepted by the spacecraft for execution. Each real-time command shall perform only one function, fully identified in the command data field.

10.1.2.1.2 ***Stored Time-tagged Single Commands*** **-** A capability to store and execute commands at the time identified by the time tag shall be provided. The time tag shall be referenced to either an absolute time or a relative time. There shall be a mechanism to read out the time-tagged contents of the command storage buffers. Stored commands shall be modifiable via ground command, including the addition and deletion of time-tagged commands. The capacity to store 25,000 commands (or the equivalent of four full 24-hour schedules) shall be provided. These schedules shall be selectable, interruptible, and uplinkable via ground command. Each time-tagged command shall perform only one function, and a command identifier shall be telemetered upon command execution.

10.1.2.1.3 ***Command Sequences*** - The capability to execute a function requiring a series of commands shall be provided. It shall be possible to call these command sequences from within other sequences to perform generic functions. At least two command sequences may be active concurrently. A command sequence identifier shall be available for telemetry upon the start of execution of the sequence, as shall the identifier for each individual command executed as part of the command sequence. Command sequences shall be executed either in real time or as stored commands.

10.1.2.1.4 ***Instrument Command Functions*** - Command functions shall be provided to support the following operations: Imager/Sounder frames, Imager/Sounder star sense, and Imager/Sounder black body calibration.

10.1.2.1.4.1 ***Imager/Sounder Frame*** - This function may be invoked by a time-tagged command executed from a storage buffer. The function will send a sequence of time-spaced commands to either the Imager or Sounder, as specified in the command data, to execute a frame. One Imager frame and one Sounder frame may be active concurrently. The command function will access frame geometry from a separate data table stored in on-board RAM. Storage will be provided for 100 different frame definitions.

10.1.2.1.4.2 ***Imager/Sounder Star Sense*** - This function may be invoked by a time-tagged command executed from a storage buffer. This function will send commands to either the Imager or Sounder, as specified in the command data, to execute a star sense. The command function will access star sense attitude data from a separate data table stored in on-board RAM. Storage will be provided for two separate tables of 2048 different Imager/Sounder stars. Only one of these tables will be active at any given time.

10.1.2.1.4.3 ***Imager/Sounder Black Body Calibration*** - This function may be invoked by a time-tagged command invoked from a storage buffer. This function will send commands to either the Imager or Sounder, as specified in the command data, to execute a black body calibration. The function will strobe the SXI to provide notification of <u>the</u> beginning <u>(deleted)</u> of the black body calibration. CCR6059 Mod 32

10.1.2.2 ***Command Data Rate, Format, and Error Detection*** - The command data transmission bit rate and format shall be as specified in Table 10.0. Additionally, the command data timing extraction mechanism shall not lose lock during the reception of the maximum possible number of continuous ones or zeros used in the command data format.

The command data format shall be as specified in Table 10.0. Detection of an incorrect decoder address by any decoder shall inhibit loading of that command, and shall not affect the existing contents of the decoder. The execution of corrupted commands shall be prevented. Loss of command carrier shall abort the execution of any real-time command and clear the command buffer. The command system shall provide an indication of command rejection.

10.1.2.3 ***Clear/Secure Command Operations*** - The spacecraft contractor shall provide an NSA-approved secure command operating mode. A command decrypter shall be associated with each command baseband chain, individually commandable, and shall have its status telemetered.

The clear text/secure text modes shall be selectable by ground command. Upon application of spacecraft bus power, the decryption key and vehicle command count (VCC) shall contain initial values of zero. New decryption keys and VCC values shall be ground commandable for each individual command decrypter. The spacecraft contractor shall propose a time-out after which both baseband chains will revert to clear mode if no command is authenticated. Failure of the command decryption hardware in one command baseband chain shall not prevent the selection of the clear text command mode for that baseband chain by the remaining command baseband chain. No command decryption hardware failure modes shall either interfere with commanding in clear text mode or reduce baseband command section redundancy in clear mode.

10.1.3 ***Telemetry Section*** - As a minimum, measurements shall be telemetered for every telemetry monitoring point on the spacecraft to minimize the reliance on ground processing to determine the spacecraft configuration. Any compound, software-computed parameters necessary to monitor the spacecraft's control behavior shall be telemetered along with their constituent telemetry monitoring points.

10.1.3.1 ***Telemetry Parameters*** - The spacecraft contractor shall provide sufficient spacecraft telemetry data channels to ensure proper control and monitoring of spacecraft health and safety, and identify anomalous conditions.

10.1.3.1.1 ***Telemetry Parameter Performance*** - The telemetered data channels shall be calibrated at the spacecraft level where possible. During spacecraft testing, applicable data channel responses shall be recorded and simultaneously presented with the subsystem test performance.

10.1.3.1.2 ***Parameter Dynamic Range*** - Each individual data channel shall have a dynamic range such that saturation does not occur during normal on-orbit operations.

10.1.3.1.3 ***Parameter Resolution*** - Each individual data channel shall be sampled at a rate sufficient to resolve parametric changes in the data being measured.

10.1.3.1.4 ***Parameter Accuracy*** - The combined parameter measurement and processing accuracy, including resolution and calibration, shall be ±3% of the full scale value for all analog signals.

10.1.3.2 ***Telemetry Time Resolution*** - Each telemetry stream data unit (packet or minor frame) shall include a time tag that allows each data channel sample within that unit to be resolved to an accuracy equal to the rate of a word in the data stream. For example, for a frame consisting of 250 eight-bit words and a data rate of 2 kbps (i.e., one frame per second), each frame would be time tagged such that the time of insertion of each eight-bit word into the frame could be resolved to the nearest 4 milliseconds (1/250 second). This requirement is not intended to place any restrictions on the instrumentation sampling rate vis-a-vis the data rate.

10.1.3.3 ***Telemetry Data Streams*** - The telemetry streams shall be defined by the spacecraft contractor, shall contain a unique spacecraft identifier, and shall be time tagged in accordance with section 10.1.3.2. The normal stream shall contain all spacecraft and instrument health and safety data needed for normal on-orbit operations activities, including SEM instrument data. Any special, spacecraft contractor-defined streams optimized for engineering uses, such as LOR, maneuvers, on-orbit storage, and safe hold mode activities, shall include SEM instrument data and SEM-associated spacecraft telemetry parameters. A capability to upload ground-defined telemetry stream definitions consisting of any combination of parameters and sample rates shall be provided. The capability to dump the contents of selected (or all) on-board computer and instrument memory and storage locations shall also be provided. Transmission of memory and table dumps, special telemetry streams, or user-defined telemetry streams shall not preclude the transmission of the normal telemetry stream. The normal telemetry stream shall also be downlinked via the MDL to provide access to SEM data to the SEC.

10.1.3.4 ***Telemetry Data Rates*** - The data rates of the DSN and CDA telemetry streams shall be as specified in Table 10.0.

10.1.4 ***T&C Functional Coverage*** - The spacecraft contractor shall provide command receive and telemetry transmit functions meeting the bit error rate (BER) requirements of Table 10.0 for the nominal ground station uplink EIRPs given in Table 10.0. Commanding (e.g., antenna switching) to achieve the command and CDA telemetry coverage requirements is not permitted.

10.1.4.1 - Normal Operations, Stationkeeping, and Housekeeping - Continuous global coverage, defined as the Earth's surface visible from the spacecraft from geosynchronous orbit, shall be available for the T&C function during all spacecraft orientations encountered in all normal operating, stationkeeping, and housekeeping modes.

10.1.4.2 - Storage and Safehold

10.1.4.2.1 - The CDA 1964 MHz telemetry link at 1 kbps must close for:

a) 100% full telemetry during the period of midnight satellite local time plus or minus 60 minutes every day during eclipse seasons.

b) An average over a spacecraft day of at least 75% full minor frame telemetry every day of the year when calculated at 1 dB design margin.

c) A worst case minimum of 50% of the full minor frames during any single spin rotation when calculated at 0 dB design margin.

10.1.4.2.2 The command link at 2 kbps must close for at least 75% of the time on the average over any spacecraft day of the year when calculated including a design margin of 1dB.

10.1.4.2.3 The DSN 2209 MHz telemetry shall be nominally off in this mode with the exception of periodic ranging operations by ground command.

10.1.4.3 *Safehold Acquisition Coverage* 10.1.4.3.1 The Command Link (2 kbps) must close at design margin (1dB) over 75% of 4pi steradian.

10.1.4.3.2 When in normal operations (non-storage) coverage via nadir antenna must be wide enough that ground operations system will capture at least 8 full useable minor frames of data after the occurrence of a credible anomaly which results in maximum spacecraft slew rate.

10.1.4.3.3 The CDA telemetry coverage requirement at 1 kbps is a minimum of 75% of 4pi steradian coverage at design margin (1dB).

10.1.5 *T&C RF Section* - The redundant DSN RF sections shall be functionally identical, with the same receiver frequencies, coherent mode transmission frequencies, and non-coherent mode transmission frequencies. No spurious signals radiated to the spacecraft or radiating within the spacecraft shall be interpreted as commands. The two CDA RF sections shall be identical with identical transmission frequencies. Either CDA telemetry transmitter shall be capable of simultaneous operation with either DSN transmitter at all times. The CDA RF section's modulation and data rate shall be as specified in Table 10.0. All DSN and CDA transmit functions and configurations shall be independently commandable ON/OFF.

10.1.5.1 *Telemetry Section EIRP* - The spacecraft contractor shall specify the DSN and CDA telemetry channel EIRPs. During all spacecraft orientations encountered in normal operating, stationkeeping, and housekeeping modes, the telemetry transmission channels shall provide a data rate of 4 kbps at a receive bit error rate (BER) of 10^{-5} with the ground receive antenna at Earth edge with a minimum 5 degree elevation angle and with a minimum link margin of 6 dB. The DSN and CDA telemetry stream receive bit error rate link margins shall account for reception by a CDAS linearly polarized antenna with a G/T of 26 dB, transmitted through the spacecraft's RHCP omnidirectional antenna system. The link calculations shall account for pointing losses and any polarization losses between the transmit and receive antennas.

10.1.5.2 *DSN RF Section Commandable Modes* - The DSN RF sections shall have the following commandable operational modes:

10.1.5.2.1 *Automatic Voltage Controlled Oscillator (VCO) Mode* - The downlink frequency shall be in phase (coherent) with the uplink signal and shall be 240/221 times the uplink frequency when the receiver is phase locked. When the receiver is not in phase lock, the downlink frequency shall automatically switch to a non-coherent auxiliary oscillator. When the receiver phase locks to an uplink carrier, the RF section shall automatically switch to the VCO within one second. The receiver lock status indicator shall be derived from a coherent AGC voltage and shall be used as the loss of lock reference. The time from loss of receiver phase lock to switch-over to the auxiliary oscillator shall include a delay of at least 200 and up to 600 msec. Carrier reacquisition without a ground sweep after a switch over to auxiliary oscillator shall be possible when the command uplink carrier frequency is maintained equal to the command receiver VCO rest frequency. Transponder receiver loop stress telemetry shall be provided that indicates an offset between the actual uplink frequency and the command receiver VCO rest frequency.

10.1.5.2.2 *Auxiliary Oscillator Mode* - The auxiliary oscillator shall be commandable ON/OFF. Receipt of an auxiliary oscillator ON command shall override the coherent AGC voltage control and force the transmitter downlink frequency to the non-coherent auxiliary oscillator frequency.

10.1.5.2.3 *Ranging Channel On/Off* - The ranging signal path to the transmitter's phase modulator shall be commandable ON/OFF. Upon receipt of a ranging channel ON command, ranging signals shall be demodulated from the uplink carrier and remodulated on the downlink carrier.

10.1.5.2.4 *Receiver Commanding* - It shall not be possible to command OFF the command RF and baseband sections.

10.1.5.3 *CDA and DSN Center Frequency and Stability* - The DSN and CDA link center frequency assignments and stability performance requirements shall be as given in Table 10.1.5.3. The DSN VCO rest frequency shall have an initial settability of less than +/- 25 ppm from the defined uplink center frequency and shall maintain a frequency stability of less than +/- 10 ppm from this initial set frequency over spacecraft life. The DSN auxiliary oscillator shall have a frequency stability better than ±15 ppm. The frequency stability of the CDA telemetry downlink shall be better than ±2.5 ppm. These frequency stability requirements apply over all satellite operating conditions and over the mission lifetime.

Table 10.1.5.3

T&C RF Performance Requirements

Center Frequency Assignments	
DSN A & B Command and Ranging Uplink	**2034.20 MHz**
DSN A & B Telemetry and Ranging Coherent Downlink	**(240/221) x Uplink (2209.086 MHz)**
DSN A Telemetry Auxiliary Oscillator Downlink	**2209.086 MHz**
DSN B Telemetry Auxiliary Oscillator Downlink	**Contractor Specified**
CDA Telemetry Downlink	**1694.0 MHz**

10.1.5.4 *DSN Acquisition and Tracking* - The tracking threshold shall not be degraded by more than 0.5 dB when the transmitter is turned on. Acquisition shall occur with the uplink signal sweeping over a frequency range of ±110 kHz around the assigned uplink frequency.

10.1.5.5 *Command Channel Performance* - The DSN RF section commandable mode performance requirements described in section 10.1.5.2 shall be met when the input signal consists of an RF carrier phase-modulated by the command subcarrier and/or ranging tones described in Table 10.1.5.5 and JPL Document 810-5. Command threshold for a DSN station is defined as the modulated uplink signal level into the transponder which produces a maximum BER of 10^{-5} for either a 1 kbps or 2 kbps omnidirectional coverage command link. The command threshold shall be met over the contractor-specified input power and the functional coverage ranges of section 10.1.4. The command channel shall perform in the clear and secure modes with the probability of any of the following events occurring of less than $1.5E^{-10}$ for discrete commands and $3.0E^{-10}$ for serial commands:

1. Missing a valid command
2. Rejecting a valid command
3. Executing a command containing an error
4. Executing a false command from a stream of random bits.

Table 10.1.5.5

DSN RF Section Ranging and Command Uplink Signal Characteristics

Uplink with single ranging tone	**0.35 to 0.70 radians peak**
Uplink with two ranging tones	**0.70 to 1.40 radians peak**
Uplink command	**0.5 to 1.3 radians peak**
Ranging plus command modulation indices	**# 2.5 radians peak**
Command data characteristics	**1 or 2 kbps and compatible with JPL 810-5**

10.1.5.6 *DSN Interfaces* - The DSN frequency telemetry and the DSN frequency command shall be as specified in Table 10.0. The telemetry, command and ranging system shall be compatible with JPL Document 810-5, Rev. D.

10.1.5.7 *CDA Telemetry Channel Characteristics* - The CDA frequency telemetry shall be as specified in Table 10.0 and shall be compatible with the SSGS NTACTS specified in section 7.1.2.

10.2 Communications

This section designates the RF end-to-end performance requirements of all communication links and the spacecraft segment used in the reception, routing, processing, and transmission of the GOES communication signals listed below in Table 10.2. The GOES communication function general requirements and transmission channel performance specifications are in section 10.2.1, with special requirements reserved for each function in sections 10.2.2 through 10.2.9.

Table 10.2

GOES Communications Signals

Function	Channel Designation
Transmit raw Imager and Sounder data	**Sensor Data Transmitter**
Relay processed imaging and sounding data, and ranging	**PDR Transponder**
Relay digital weather facsimile	**WEFAX Transponder**
Relay Emergency Managers Weather Information Network data	**EMWIN Transponder**
Relay data collection platform interrogation signals	**DCPI Transponder**
Relay data collection platform reports	**DCPR Transponder**
Relay search and rescue beacon transmissions	**SAR Transponder**
Transmit SXI, dynamic interaction attitude data, and CDA telemetry	**MDL Transmitter**

Unless otherwise specified, the performance requirements of section 10.2 shall be met (1) under all available combinations of routing path and redundant hardware available to the transmission channel, including all coverage provided by the antenna subsystem; (2) over all spacecraft operating conditions of normal operations, station keeping, and housekeeping, including eclipse and throughout the mission lifetime, however excluding yaw flip maneuvers for linearly polarized transmission channels. (3) over all combinations of telemetry, tracking, command, and on-orbit communication functions which shall operate without degradation to or interference with each other. (4) Excluding periods when the command uplink EIRP from any transmitter is at the CDA frequency and is outside the CDA transmitter EIRP range shown in Table 10.0.

10.2.1 *General Requirements* - Refer to section 13 for the definitions of various communication system terms used herein.

10.2.1.1 *Commandable Emissions ON/OFF* - It shall be possible to turn on or off, by ground command, each individual transmission channel and each electronic component in the communication subsystem. There shall be no limitation to this capability at any time during the orbital design life or to the duration and frequency of the off or on times.

10.2.1.2 *Dynamic Range* - Each communications function transmission channel shall meet all specifications when its input signal is within the dynamic range specified in Table 10.0 and need not perform to specification when the signal is not within the dynamic range. However, the rest of the spacecraft, including any combination of the communication system functions, shall perform to specification.

10.2.1.3 *Transponder Reference Point* - Communication subsystem input and output points refer to the receive and transmit antenna terminals, respectively. Unless otherwise stated, all signal input levels in this specification shall be referred to isotropic (dBmi), which is equal to the transponder input if preceded by a 0 dB gain antenna.

10.2.1.4 *Transponder Channel Overdrive Capability* - The communication system transponder relay channels shall be designed such that each transponder channel can withstand, without subsequent performance or lifetime degradation, prolonged operation with single or multicarrier RF illumination up to -60 dBmi. The frequency of the illumination signals may be anywhere within the transponder bandwidth. Communication system performance specifications need not be met during overdrive conditions; however, the transponder shall be operational upon cessation of the overdrive condition.

10.2.1.5 *Antenna Polarization* - Linear polarized antennas on the spacecraft shall have the polarization aligned along the earth's spin axis (north-south). Right hand circular (RHC) polarization shall be according to the standard IEEE definition.

10.2.1.6 *Downlink Frequencies and Stability* - The on-orbit functions shall use the uplink and downlink frequencies given in Table 10.2.1.6. The short-term frequency stability of all frequency conversion oscillators shall be better than $\pm 10^{-9}$ when measured over a 0.25-second interval, and the S-band and L-Band downlink long-term frequency stabilities shall be better than ±3.0 parts per million (ppm) referenced to the downlink center frequencies. The UHF downlink long-term frequency stability shall be better than ±5.0 ppm referenced to the downlink center frequencies. The stability of the SAR transponder frequency translation to baseband shall be better than ±2.0 ppm referenced to the uplink frequency.

Table 10.2.1.6

On-Orbit Communication Function Uplink and Downlink Center Frequencies

Function	Uplink (MHz)	Downlink (MHz)
PDR Transponder	**2027.700**	**1685.700**
WEFAX Transponder	**2033.000**	**1691.000**
EMWIN Transponder[1]	**Contractor Spec**	**Contractor Spec**
DCPI Transponder:		
DCPI Spare	**2034.8875**	**468.8125**
DCPI East	**2034.9000**	**468.8250**
DCPI West	**2034.9125**	**468.8375**
DCPR Transponder:		
Domestic	**401.900**	**1694.500**
International	**402.200**	**1694.800**
SAR Transponder		
Narrowband	**406.025**	**1544.500**
Wideband	**406.050**	**1544.500**
Sensor Data	**----**	**1676.000**
MDL Data	**----**	**1681.478**

[1]The EWIN Uplink Frequency shall be between 2030.1 MHz and 2034.8 MHz and the downlink frequency shall be between 1688.0 MHz and 1693.9 MHz. The contractor may, if desired, use a translation frequency other than that used for the PDR and WEFAX transponders.

10.2.1.7 ***AM/PM Conversion*** **-** The AM/PM conversion of any transmission channel shall not exceed 5^{o} per dB, <u>expect for the SAR channel</u>.

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Table 10.2.9.3

<u>SAR Receiver Bandwidth Characteristics</u>

Operating Mode	Center Frequency (MHz)	Bandwidth (kHz)	Relative Response (dB)
Wideband	406.05	60 Minimum	-1
	406.05	79 Minimum	-3
	406.05	130 Maximum	-20
Narrowband	406.025	12 Minimum	-1
	406.025	19 Minimum	-3
	406.025	40 Maximum	-20

10.2.1.8 ***Link BER Performance*** **-** Each communication subsystem function link performance shall sustain the minimum digital transmission quality of Table 10.2.1.8-1 with global coverage and the three minimum statistical link margins (see *CCSDS Recommendations for Radio Frequency and Modulation Systems*, CCSDS 401) shown in Table 10.2.1.8-2 for all link parameters in Table 10.0. The G/T parameters, the Receive System Losses, and the BER in Table 10.0 and Table 10.2.1.8-1 are the "Design Values" for use in the statistical link margin calculation methodology. The EIRP values for PDR, WEFAX, EMWIN, and DCPI are nominal values that may be increased only until the minimum dynamic range requirement of 10.2.1.1 is realized. The EIRP values for DCPR and SAR are the "Design Values" for use in the statistical link margin calculation methodology.

With the exception of the SAR channels, the spacecraft contractor shall determine the minimum transmit EIRP of each communication channel required to meet the link performance requirements of this section when transmitting to an earth station at Earth edge (0^{o} elevation angle) for the Adverse Tolerance, to an Earth station with a 5^{o} elevation angle for the Design Value and an Earth station with a 15^{o} elevation angle for the Favorable Tolerance. For each link, the link design margins given in Table 10.2.1.8-2 shall include the contribution of all signal-to-noise ratio, propagation, polarization, and satellite channel waveform distortion losses. (Sentence Deleted). A 0.5 dB pointing loss Design Value shall be allowed for the Suitland PDR and MDL receive antennas since they have no autotrack capability. The same Design Value pointing loss shall also be allowed for the CDAS 18-meter GOES antennas (transmit and receive) since they use step trackers. Polarization loss Design Value of 0.2 dB shall be used for all linear ground and satellite antenna combinations.

10.2.1.8.1 ***WCDAS Uplinks*** **-** In normal operations the WCDAS Uplinks the DCPI, PDR, WEFAX, EMWIN, and Telecommand signals. The signals are simultaneously transmitted through a single klystron power amplifier.

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Table 10.2.1.8-1
Digital Transmission Channel Performance Specifications

Function	BER	Receive System Losses (dB)	Channel Modulated Signal Structure	Adjacent Channels for BER Test
PDR	**10^{-6}**	**2.3**	**2.11 Mbps, NRZ-S, BPSK (Current Instruments)** **2.374 Mbps, NRZ-S, BPSK (Adv. Instruments)**	MDL and WEFAX
WEFAX	**10^{-8}**	**2.3**	**293 ksps, NRZ-M, BPSK, Coded**	**PDR and EMWIN**
EMWIN	**10^{-2}**	**2.3**	**25 kbps, NRZ-L, BPSK, Un-coded**	**WEFAX**
DCPI	**10^{-5}**	**2.0**	**100 bps, Biö-L, ±60° PSK**	**-**
DCPR	**10^{-6}**	**2.0**	**100 bps, Biö-L, ±60° PSK** **(Deleted)**	**-**
DCPR	**Not applicable**	**Not applicable**	**300 bps, Trellis Codulated 8PSK** **1.2 kbps, Trellis Codulated 8PSK**	**Not applicable**
SAR	**10^{-5}**	**2.0**	**400 bps, Biö-L, 1.1 rads peak, 500 msec pulse every 50 seconds**	**-**
Sensor Data **Imager** **Sounder**	 **10^{-8}** **10^{-8}**	**2.3**	**2.62 Mbps (I), 40 ksps (Q), NRZ-S, OQPSK (Current Instruments)** **2.934 Mbps, 720 ksps, NRZ-S, OQPSK (Adv Instruments)**	**MDL**
MDL	**10^{-8}**	**2.3**	**Contractor Spec ksps, NRZ-L, QPSK**	**SD and PDR**

Table 10.2.1.8-2
Statistical Link Margin Performance Specifications at Earth Edge

Link Specification	Minimum Link Margin (dB)
Design +2ó	**> 2.00**
Design	**> 1.00**
Design -2ó	**> 0.00**

The minimum EIRPs of the DCPR and SAR downlinks are specified in Table 10.0 because of the wide range of ground receive systems with different capabilities. The DCPR EIRP is specified at 3 dB above the GOES I/M EIRP to accommodate future 300 bps and 1.2 kbps platforms.

10.2.1.9 ***Power Flux Density Limits*** - No transmission channel shall exceed earth surface power flux density (PFD) limits specified in the International Telecommunications Union (ITU) regulations by more than 1 dB when accessed by the corresponding modulated signal spectrum of Table 10.2.1.8-1.

If the spacecraft contractor determines that any transmission channel requires a higher than allowed EIRP given the performance requirement and the ground segment characteristics in Table 10.0, the contractor shall notify NASA.

Table 10.2.1.9
Power Flux Density Limits (dBW/m²/worst-case 4 kHz band)

Frequency Band (MHz)	Angle of Arrival		
	0° - 5°	5° - 25°	25°-90°
1670 - 1700	-154.0	-154.0 + 0.5(angle - 5°)	-144
1544 -1545	No Limit	No Limit	No Limit
460 - 470	-152.0	-152.0	-152.0

10.2.1.10 ***Spurious Emission and Intermodulation Product Requirements***

10.2.1.10.1 ***In-band Spurious and Intermodulation Products*** - Each transmission channel's spurious emissions shall be # -60 dB below the unmodulated downlink carrier over 1.5 times the channel 3 dB bandwidth in the presence of all other modulated uplink (within the dynamic range) and downlink signals.

The ratio of the carrier to third-order intermodulation products within the 3 dB bandwidth of each transmission channel, single or multiple access, shall be no less than 40 dB for common input sections and no less than 30 dB for common output sections at the spacecraft output, under the following conditions:

1. All communication signals in any shared channel bandwidth shall be unmodulated carriers 3 dB below their dynamic range maximum.
2. All combinations of frequencies where any communication signal is varied within the transponder bandwidth such that at least one third-order intermodulation product falls within any transmission channel.

10.2.1.10.2 ***Out-of-band Spurious*** - For any and all frequencies in the RF spectrum, excluding the frequency bands 1660.0 to 1697.0 MHz, 1544.0 to 1545.0 MHz, and 465 to 471 MHz, the total power resulting from the sum of all spurious signal, including the second harmonics of the communication signals radiated from the spacecraft shall not exceed - 20 dBmi in any 4 kHz frequency segment.

10.2.1.10.3 ***Power Spectral Density Line Spurious*** - Discrete signal lines contrary to the theoretical modulated power spectral density for telemetered NRZ OQPSK and QPSK data shall be #-<u>25</u> dBc. CCR6014 Mod 32

10.2.1.10.4 ***Radio Astronomy Band Spurious*** - For the bandwidth from 1660 MHz to 1670 MHz, the power flux density measured anywhere on the Earth's surface shall not exceed:

Line: -201 dBW/m², measured in a 20 kHz bandwidth
Continuum: -213 dBW/m², measured in a 10 kHz bandwidth.

10.2.1.11 ***Out-of-band Rejection*** - The rejection by any transmission channel outside of the bandwidths 401.50 to 406.30 MHz and 2018 to 2042 MHz shall be > 30 dB.

10.2.1.12 ***Phase Noise*** - The RMS phase noise on all communication function downlink carrier frequencies shall not exceed:

1. A maximum of 5^{o} on the DCPI and DCPR channels, measured using a phase locked loop (PLL) with a 10-Hz single-sided noise bandwidth (B_L) over the frequency range from the center frequency to 1 kHz maximum.
2. A maximum of 5^{o} on all other channels, measured using a PLL with a B_L # 30 Hz over the frequency range from the center frequency to the 3 dB bandwidth point.

10.2.2 ***Sensor Data Transmission Requirements*** - The Sensor Data transmission channel shall receive the GFE Imager and Sounder data streams, and independently and asynchronously modulate each data stream in phase quadrature (OQPSK) with unbalanced carrier power. In the absence of either data stream, the transmission channel EIRP shall not be reduced and the BER performance of the remaining channel shall not be degraded.

The GFE Imager and Sounder instrument electronics modules generate a PN code whenever they are powered on. This PN code is added to the instrument data before the data stream is passed to the Sensor Data transmission channel. ~~For BER testing the instruments can be commanded to a state in which they emit a known data pattern.~~ The PN code generator is defined in the instrument ICDs. With the instrument electronics powered off, the Sensor Data transmission channel shall emit a CW carrier for use in EIRP and long-term frequency stability testing. CCR 6003 Mod 24

10.2.2.2 ***Modulation Characteristics*** - The Imager and Sounder data shall be modulated in the in-phase (I) and quadrature (Q) channels, respectively; and the carrier power ratio of the Imager to the Sounder data shall be 4:1. A positive-going voltage at the modulation input shall cause the phase of the RF carrier to advance (become more positive).

10.2.3 ***MDL Transmission Requirements*** - The MDL transmission channel shall multiplex the Imager Servo Error, the Imager IMC Analog data, Sounder Servo Error, SXI data, Imager ADS, SXI ADS, IOO data (from one or two instruments), normal mode Telemetry, and Dwell mode Telemetry and any other contractor specified data. The output data shall be NRZ-L formatted and shall include at a minimum a parity bit in the header to allow for bit error detection. This data stream shall then be balanced QPSK modulated. Refer to the Imager, Sounder, SXI Interface Control Documents (ICDs), and the IRD for the IOO for data rate and data format information. The spacecraft contractor shall format the data stream to resolve the four-fold phase ambiguity of the demodulated data. The MDL transmission channel shall also have a ground commandable mode in which the downlink carrier is QPSK modulated only by a PN sequence for use in BER testing. Futhermore, the MDL transmission channel shall have a ground commandable mode through which a CW RF carrier can be emitted for EIRP and long-term frequency stability testing. CCR6043 Mod 36

Table 10.2.3
Parameters and Rates

Parameter	Sample Rate
Frame sync	Once per frame
Imager servo error	1000 Hz, each N/S and E/W
Imager IMC analog data	1000 Hz, each N/S and E/W
Sounder servo error	400 Hz, each N/S and E/W
SXI ADS data	800 Hz each axis
Optical bench ADS data	800 Hz each axis
SXI image data	100 KBPS
LM image data	100 KBPS
Normal mode S/C telemetry data	4 KBPS
Dwell mode S/C telemetry data	4 KBPS

10.2.3.2
Modulation Characteristics - The MDL multiplexer data stream to the MDL modulator shall be continuous.

10.2.4 ***Processed Data Relay Transponder Requirements***

10.2.4.1 ***No Access Output Power*** - To enable CDAS ground system time delay calibration for PDR ranging, the PDR transponder EIRP shall not exceed +10 dBm with a carrier uplink of 0.25 watts, . Compliance with this requirement shall be demonstrated during ground tests with an input test cable providing a 290 K source temperature and no signal.

10.2.4.2 ***Time Delay*** - The PDR transponder time delay shall not exceed 1400 ns, when measured with a digital PN sequence, and shall not vary more than ±25 ns over all spacecraft environments.

10.2.5 ***Digital WEFAX Transponder Requirements*** - The digital WEFAX signal defined in Tables 10.2.1.10 and 10.2.1.12 includes the use of concatenated convolutional rate ½, constraint length K = 7 inner and Reed-Solomon 223, 255 outer code.

10.2.5.1 ***Baseband Pulse Shaping*** - The digital WEFAX implementation includes the use of a Nyquist raised-cosine channel filter in cascade with an x/sin x-shaped aperture equalizer for shaping the NRZ pulses prior to RF transmission. The CDAS transmit channel characteristic shall have a roll-off of 1.0 and an apportionment of 0.5. The out-of-band attenuation at (1 + 1.0)146.5 ksps shall be $ 30.0 dB.

10.2.5.2 ***Sideband Regrowth*** - Sideband regrowth due to satellite channel non-linearities shall not produce sidebands greater than -18 dB referenced to the modulation main lobe.

10.2.6 ***EMWIN Transponder Requirements*** - The EMWIN link shall provide a minimum BER performance of 10^{-2} at a data rate of 25 kbps with no forward error correction coding, NRZ-L formatted and BPSK modulated, in compliance with Table 10.2.1.8-1. The EMWIN transponder EIRP shall be the maximum allowed by the PFD limits of subsection 10.2.1.9.

10.2.7 ***DCPR Transponder Requirements -*** The DCPR transponder is frequency division multiple accessed (FDMA) with two 400 kHz bands, termed the domestic and international bands, respectively. Each band is further subdivided into 200 1.5-kHz and 33 3-kHz channels. Data collection platforms operating at 100 and 300 bps use 1.5 kHz channels, and 1.2 kbps platforms will be assigned two contiguous channels (3 kHz). Table 10.2.7 shows the DCPR frequency bands, transmission channel bandwidths, and noise bandwidths.

Table 10.2.7

DCPR Frequency Bands and Noise Bandwidths

Band	Center Frequency (MHz)	FDMA Bandwidth (MHz)	Max Noise Bandwidth (kHz)
1	401.9000	401.7000 - 402.1000	500
2	402.2000	402.0000 - 402.4000	500

10.2.7.1 ***Link Margin Requirements*** **-** The DCPR transmission channel shall meet the minimum link margin requirements for the 100 bps BPSK platform type in any assigned channel bandwidth in either frequency band of Table 10.2.7, and in any single or multiple access combination and quantity of platform types. The 8PSK platform types shall meet the EIRP and G/T requirements of Table 10.0. CCR6017C Mod32

10.2.7.2 ***Dynamic Range*** **-** The DCPR transponder shall meet the performance requirements over an input dynamic range from no signal to -100 dBmi, where the -100 dBmi corresponds to the presence of 233 simultaneous platform reports.

10.2.7.3 ***Automatic Gain Control*** **-** The transponder shall have an AGC to regulate the total power (signal plus noise) in the transponder. When driven by a step signal tone at the band center frequency corresponding to a step from no drive to any and all drive levels up to the dynamic range maximum, the amplitude and phase responses of the repeater shall be within ±0.5 dB and $\pm 3^{o}$ of steady state values within 15 msecs of the start of the burst and shall remain within those limits until the end of the burst 0.6 seconds later. The start of the drive burst is defined as the time at which the transponder input power is 3 dB above the channel noise.

10.2.7.4 ***Noise Power Ratio*** **-** The ratio of the noise power density to the power density measured in the center slot at the channel output is defined as the noise power ratio. The noise power ratio shall be ≥ 20 dB with wideband Gaussian noise at the spacecraft input extending over the full transponder bandwidth measured in a 1 kHz center slot at the downlink frequency.

10.2.7.5 ***CDA Telemetry Channel Interference*** **-** With no DCPR signal input, the DCPR channel noise power in the CDA telemetry channel shall be at least 30 dB below the telemetry power as seen by a ground station in the far field.

10.2.7.6 ***In-band Spurious Outputs*** **-** The in-band spurious outputs of the DCPR transponder shall be ≤ -160 dBmi at the transponder input with: 1) no platform report signal at the input, and 2) a -100 dBmi input signal simulating 233 simultaneous reports.

10.2.7.7 ***DCPR Commandable Gain Steps*** **-** A capability to command monotonic DCPR EIRP reductions shall be provided. At least fifteen gain steps, each no greater than 1.5 dB, shall be provided, one being 0 dB (maximum EIRP). It shall be possible to command EIRP increases and reductions in single steps, and a command shall be provided to reset the EIRP to maximum. In addition, there shall be no limit on the number of times the EIRP is adjusted over the mission life. The transmission channel linearity requirement shall be met at all gain step settings. Admin. Chg. Correction to Mod 32

***Frequency Response* -** The frequency response of the DCPR transmission channel shall not fall more than 1.5 dBc from the level at center frequency over the band of 401.7 to 402.1 MHZ and the center frequency over the band of 402.0 to 402.4 MHZ. CCR6017C Mod 32

10.2.7.8 ***Phase Response*** - The phase response of the DCPR transmission channel shall not vary more than 60 degrees over the band of 401.7 to 402.1 MHZ and over the band of 402.0 to 402.4 Mhz.

10.2.8 ***DCPI Transponder Requirements*** - The DCPI transponder shall meet the minimum link margin requirements when accessed by one of the carriers with center frequencies shown in Table 10.2.1.6. A capability to command a monotonic reduction in the DCPI transponder EIRP in steps not greater than 1.25 dB down to at least 3 dB below the PFD limit shall be provided, with no limit on the number of times the EIRP is adjusted over the mission life. Command capability will be increment, decrement, and reset to maximum EIRP.

10.2.9 ***SAR Transponder Requirements***

10.2.9.1 ***Commandable Modes*** - The SAR transponder shall have the following commandable modes and meet its minimum link margin requirements in any combination:

1. Narrowband or wideband channel selection.
2. Automatic level control (ALC) or fixed gain selection.

10.2.9.2 ***Frequency Band Assignments and Frequency Stability*** - The SAR frequency assignments shall be as shown in Table 10.2.1.6. The SAR transponder baseband conversion frequency stability averaged over a 600-second time interval shall be better than ±0.5 Hz.

10.2.9.3 ***Bandwidth Characteristics*** - The SAR transponder shall meet minimum link margin performance requirements over a central 5 kHz bandwidth. The channel bandwidth characteristics shall be as shown in Table 10.2.9.3. The group delay slope within the 1 dB bandwidth specified shall not exceed 13 µsecs per kHz when measured over any 4 kHz band.

Table 10.2.9.3
SAR Receiver Bandwidth Characteristics

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Operating Mode	**Center Frequency (MHz)**	**Bandwidth (kHz)**	**Relative Response (dB)**
Wideband	406.05 406.05 406.05	60 Minimum 79 Maximum 130 Maximum	-1 -3 -20
Narrowband	406.025 406.025 406.025	12 Minimum 19 Maximum 40 Maximum	-1 -3 -20

10.2.9.4 ***Modulation Characteristics*** - The SAR transponder output shall phase modulate the received and downconverted baseband noise process, where 0 Hz corresponds to 406.0 Mhz.

10.2.9.4.1 ***Modulation Input/Output Polarity*** - A positive-going voltage at the modulation input shall cause the phase of the RF downlink carrier to advance (become more positive).

10.2.9.4.2 ***Modulation Index*** - The transponder modulation index shall be such as to provide between 2 and 5 dB of carrier suppression on orbit and the carrier suppression shall remain within ± 0.25 dB over any 20 minute interval, with the noise power as seen on orbit, in all transponder commandable modes and configurations. The spacecraft contractor shall analytically determine the system temperature difference (the ambient channel noise power difference) between the on-orbit environment and the integration and test environment. There shall be no measurable spurious

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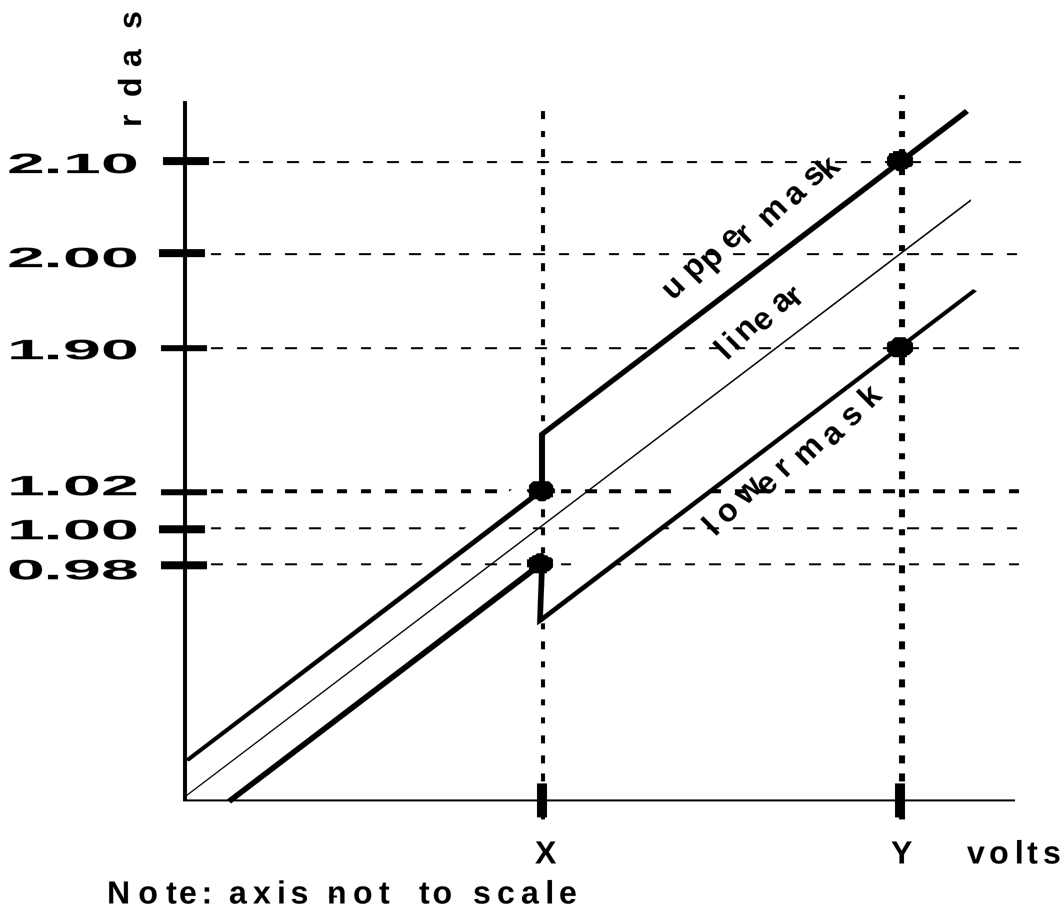
modulation of the SAR repeater due to the operation of any spacecraft function as measured at the transponder input that exceeds -175 dBmi.

10.2.9.4.3 ***Modulator Amplitude Line -*** The amplitude non-linearity of the phase modulator shall not exceed ±0.02 radians deviation from linear between zero and 1 radian peak modulation index and shall not exceed ±*0.10* radians deviation from linear between 1 and 2 radian peak modulation index as shown in Figure 10.2.9.4.3-01. CCR4059 Mod 24

Figure 10.2.9.4.3-01
SAR Modulator Amplitude Linearity Requirements



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10.2.9.4.4 ***Modulation Limiter*** - The transponder shall have a limiter preceding the phase modulator to limit the downlink instantaneous modulation index due to voltage fluctuation noise to 2 radians of peak deviation.

10.2.9.4.5 ***Transmitter Output Spectrum*** - All harmonics of the transmit frequency shall be at least 50 dB below the unmodulated carrier.

10.2.9.4.6 ***In-band Spurious Signals*** - The in-band spurious signals generated by the spacecraft shall be less than -175 dBmi at the transponder input. Compliance to this requirement shall be demonstrated according to the spurious emission testing requirements of Paragraph 3.3.8.4 of Document S-415-23. CCR4059 Mod 24

10.2.9.5 ***Selectable Gain Mode Performance*** - As used herein, the gain is that between the spacecraft input and the point before the modulation limiter.

10.2.9.5.1 ***ALC Mode*** - The ALC mode, when selected, shall regulate on the signal plus noise power in the transponder channel to 15 dB above the channel noise power. The ALC time constant shall be within 25 msecs to 35 msecs.

10.2.9.5.2 ***Fixed Gain Mode*** - When the fixed gain mode is selected, the ALC mode shall be disabled and the receiver gain shall be such that the modulation index of section 10.2.9.4.2 is achieved.

10.2.9.6 ***Linearity*** - The SAR transponder, from the input antenna to the limiter input point, shall meet the general linearity requirement of section 10.2.1.10. Two equal-level test tones loading the channel at the dynamic range maximum, spaced anywhere in frequency within the wide or narrow bands to minimize ALC interaction shall not produce third-order intermodulation distortion $\geq$ 30 dB below either test tone in the ALC mode or $\geq$ 15 dB below either test tone in the fixed gain mode.

10.3 Attitude Control

10.3.1 ***General Requirements*** - The ACS performance shall support all GOES N-Q mission requirements.

10.3.2 ***Design Margins***

10.3.2.1 ***Rigid Body Stability*** - The rigid-body stability margins for all ACS operational modes, over the mission life, shall be as follows:

1. Gain Margin > 6 dB (upper and lower, if applicable)
2. Phase Margin > 30 degrees

10.3.2.2 ***Flexible Body Stability***

10.3.2.2.1 ***Damping Ratio*** - For the purpose of analysis and simulation, the damping ratio of all flexible body modes shall be assumed to be no greater than 0.1%, unless analysis or test data demonstrate otherwise.

10.3.2.2.2 ***Gain Stabilization*** - ACS control laws and compensation shall gain-stabilize all flexible-body modes, except in special cases where gain-stabilization is shown to be a severe design driver. The peak amplitude of each gain-stabilized flexible-body mode shall not exceed -12 dB in the ACS open-loop frequency response.

10.3.2.2.3 ***Phase Stabilization*** - Any/all flexible-body modes which do not meet the gain-stabilization requirement of section 10.3.2.2.2 shall have phase margin of at least 60° over a modal frequency variation of ±25%, with worst-case time delays included.

10.3.2.3 *Momentum Management* - Total momentum storage capacity of the normal operating complement of reaction and/or momentum wheels shall be at least three times the worst-case accumulated momentum due to environmental torques. The capability to quickly restore the momentum storage capacity of the system shall be provided for all on-station wheel control modes of

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operations (reference sections 4.2.2, 4.2.5 and 4.2.6). This capability shall be initiated from the ground using a single command sequence (Reference 10.1.2.1.3) which does not require any real-time operator or ground system calculations or decisions. Upon initiation of this capability, nominal INR performance (reference sections 4.1.2 through 4.1.4.2.2) shall be restored within 24 hours provided initiation occurred during Normal Operations Mode (reference section 4.2.2). The momentum management system, operations concept, and actuator selection shall be consistent with these requirements.

10.3.3 ***Failure Detection and Correction (FD&C)***

10.3.3.1 ***General*** **-** The ACS FD&C algorithms shall monitor the status and performance of each ACS component and report failures/anomalies to the ground. Provisions shall be made for autonomous corrective action as described in section 4.5.

10.3.3.2 ***Enable/Disable*** - It shall be possible to enable or disable individual FD&C elements by ground command.

10.3.4 *Safe Hold Mode*

10.3.4.1 *General* - The spacecraft shall include an independent safe hold mode which shall acquire and maintain a thermally safe and power positive spacecraft attitude for an indefinite period of time, independent of the initial attitude. The safe hold attitude shall be consistent with maintaining the health and safety of the spacecraft and instruments. The design and operation of the safe hold mode is to be simple and robust. The safe hold mode shall be autonomous, requiring minimal or no ground intervention to maintain the spacecraft in this mode. Safe hold mode control shall be independent from primary hardware and software to the maximum practical extent.

10.3.4.2 *Enable/Disable* - The safe hold mode shall be capable of being enabled, disabled, or activated by ground command.

10.3.4.3 *Thrusters* - The safe hold mode shall not autonomously operate thrusters having a maximum thrust > 0.01 Newtons. The safe hold mode shall provide the capability for operating thrusters > 0.01 Newtons by ground command only.

10.3.4.4 *Telemetry* - The safe hold mode shall provide adequate telemetry to determine the health and safety of the spacecraft and the operation of safe hold, independent of the status of the computer/ processor.

10.4 Propulsion

10.4.1 ***General Requirements*** - The spacecraft shall provide the propulsion and control capability required to maneuver from the deployment orbit to geostationary orbit considering 3ó dispersion for all error sources from separation through end-of-life. The propulsion subsystem shall provide sufficient propellant to accomplish all maneuvers during the lifetime of the spacecraft, including all modes of operation, as detailed in section 4.2.

10.4.2 ***Plume Impingement*** - The propulsion subsystem shall prevent damage to GOES N-Q surfaces or contamination to instruments due to plume impingement.

10.4.3 ***Interlocks*** - The design of the propulsion subsystem shall provide the interlocks required to preclude dry firing of the thrusters. The design shall also provide for the ability to override the interlocks.

10.4.4 ***Valve Telemetry*** **-** Thruster valve operation telemetry indications shall be based on ACE software commanded thruster activity. CCR 4138 Mod 24

10.5 Power and Electrical

10.5.1 ***Energy*** **-** The power subsystem's solar array and batteries shall provide sufficient energy to support any operating mode within the requirements of this specification throughout the spacecraft lifetime.

10.5.2 ***Load Control*** **-** The spacecraft shall be capable of connecting and disconnecting each load individually by command. Essential or critical loads may be hard-wired to the spacecraft bus. It shall be impossible to disconnect or otherwise disable the command function. (Deleted sentence) Some short-term current sharing with the battery for selected peak power conditions is allowed if approved by NASA. CCR6049 Mod36

10.5.3 ***Arm and Safe Functions*** **-** Arm and safe functions meeting the safety and reliability requirements established by range safety shall be included.

10.5.4 ***Solar Array*** **-** The solar array shall be capable of meeting all operational power requirements without power sharing with the batteries during any normal sunlight operational mode throughout the specified spacecraft lifetime. Some short-term current sharing with the battery for selected peak power conditions is allowed if approved by NASA.

10.5.4.1 ***Test Requirements*** - The GOES N, O, P, and Q flight panels shall be tested to the protoflight test levels specified in Table 10.5.4.1. However, if any of the GOES N, O, P, or Q solar panels are identical in size, design, and construction to a solar panel previously qualified to GOES N, O, P, and Q thermal, environmental, mechanical, and electrical requirements, the GOES N, O, P, or Q flight panels shall be tested to the acceptance test levels specified in Table 10.5.4.1. For GOES N, O, P, and Q, life cycle test coupons consisting of a minimum of 3 strings of cells shall be constructed using the flight panel fabrication procedures. The life cycle test coupons shall include all the design and construction characteristics of their respective flight panel and may only vary in size. A minimum of 800 thermal cycles at qualification temperatures shall be performed on the life cycle test coupons for GOES N, O, P, and Q. The life cycle test coupon can be eliminated if a previous life cycle test coupon exists which is identical in all the design and construction characteristics of the respective GOES N, O, P, or Q solar panel and the number of cycles and temperature ranges envelope those of GOES N, O, P, and Q. A thermal cycle test and a thermal-vacuum cycle test shall be conducted on all the populated GOES N, O, P, and Q flight panels. The thermal cycle test shall consist of a minimum of eight thermal cycles and the thermal-vacuum cycle test shall consist of a minimum of four thermal-vacuum cycles as defined in 8.5.3.2 of this spec. The thermal cycling test shall be conducted at the component level and the thermal-vacuum cycling test shall be conducted at the spacecraft level as defined in 8.5.3.2. In the event of a conflict between sections 8.5.3.2 and 10.5.4.1, section 10.5.4.1 shall govern. During each solar array thermal-vacuum cycle test, the outgassing rates of all the flight panels shall be measured with temperature-controlled quartz crystal microbalances (TQCM) to verify that the solar array outgassing requirement is met. If the solar array outgassing requirement is not met, a corrective bake-out phase will be assessed and implemented. For GOES N, obtain at least 12 GOES N solar cells and verify that the average Solar Absorptance is consistent with the established MAT range.

Table 10.5.4.1 GOES N, O, P, and Q Solar Panel
Protoflight And Acceptance Test Levels

Test	Protoflight Test Levels	Acceptance Test Levels
Thermal Cycling	15 /C above the high temperature extreme of the MAT range 15 /C below the low temperature	10 /C above the high temperature extreme of the MAT range 10 /C below the low temperature

	extreme of the MAT range	extreme of the MAT range
Thermal - Vacuum Cycling	15 /C above the high temperature extreme of the MAT range **Cold Extreme:**s -150 /C	5 /C above the high temperature extreme of the MAT range **Cold Extreme:** -150 /C
Structural Loads	1.25 x Limit Load	1.0 x Limit Load

10.5.5 ***Batteries***

10.5.5.1 ***General Requirements*** - The battery(s) shall be fully capable of performing its function throughout spacecraft launch modes, transfer orbit modes, on-orbit storage, and operational lifetime. Flight batteries shall not be used during routine ground test and integration and shall not be installed prior to 90 days before the scheduled launch.

10.5.5.2 ***Depth-of-Discharge*** - The battery(s) shall have sufficient capacity to operate the spacecraft on-orbit through all eclipses up to 72 minutes, with the depth of discharge not to exceed 75%.
For launch ascent, lunar eclipse, on-orbit storage, or any single event at reduced power demand, the depth of discharge shall not exceed 75%.

A capacity check, pressure versus state-of-charge, calibration, reconditioning, or an effective alternative, spacecraft contractor proposed/NASA approved, shall be performed on the battery(s) prior to each eclipse season without removing all battery capacity from the spacecraft power bus.

10.5.5.3 ***Reliability*** - The battery(s) shall be capable of meeting all performance requirements after sustaining any single credible cell failure.

10.5.5.4 ***Charging*** - The battery charge control system shall be capable of recharging the battery(s) to a full state-of-charge at least once during each 24-hour period without over stressing the battery(s).

10.5.5.5 ***Test Requirements*** - One battery shall be qualification tested and all others shall be acceptance tested to demonstrate they are sufficiently reliable for the spacecraft mission. If an identical battery to the GOES N-Q battery has been previously qualified to GOES N-Q thermal, environmental, mechanical, and electrical requirements, battery qualification does not have to be performed for GOES N-Q. The qualification battery may be used for an integration and test battery. However, the qualification battery shall not be used for flight.

10.5.5.6 ***Telemetry*** - Battery cell voltage telemetry from each of the battery cells of the GOES N-Q flight batteries shall be provided to the GOES N-Q spacecraft and routed to GTACS. This telemetry is for display purposes only and need not be incorporated in any procedures or special ground processing.

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10.5.6 ***Grounding*** - Grounding shall be accomplished to minimize radiated and conducted noise, including common mode noise, and shall simultaneously provide electrostatic discharge (ESD) protection. Primary power (defined as the input to a DC to DC converter) shall be isolated from secondary power (defined as the output of a DC to DC converter). Primary power returns shall be connected to chassis at one point (single point ground concept) with a minimum impedance connection. Primary current shall not flow on secondary returns , and secondary power shall not flow on primary returns. Moreover, DC currents shall not flow through the chassis.
Secondary power returns shall be referenced to chassis with an impedance sufficient to meet the requirements of section 8.4, including the common mode noise requirements of section 8.4.11. The chassis shall be used as a zero signal reference plane for RF, fast rise time signals, and sensitive sensor

front ends (multipoint ground concept). All interfaces between boxes shall be designed for compliance with the common mode noise requirements of 8.4.11.
Where converters are cascaded (the output of one converter is connected to the input of another), a new single point primary power ground reference (node) may be selected to establish a minimum impedance reference.

10.5.7 ***Short Circuit Prevention*** - The GOES N-Q spacecraft shall prevent a short circuit in any component from damaging any other component.

10.5.8 ***Non-fused Power Lines*** - The GOES N-Q spacecraft non-fused power lines shall be double-insulated.

10.5.9 ***Power Regulators/Power Supplies*** - All power regulators/power supplies shall be stable. They shall not oscillate when operated with any operational loads and subjected to any operational environmental conditions contained in this specification. The power regulators/power supplies shall also be stable and not oscillate when operated at any load conditions between 20% above maximum operational load power and 20% below minimum operational load power. The power regulators/power supplies shall have a phase margin of better than 45^o and a gain margin of better than 12 dB.

10.6 Thermal Control

10.6.1 ***General Requirement*** - The spacecraft and instruments shall meet the requirements of this specification during all encountered thermal environments (system level test, launch through transfer orbit, synchronous orbit, and on-orbit storage mode).

10.6.2 ***Design Requirement*** - The thermal design shall maintain the spacecraft and instrument subsystems and components within their MAT limits during system level thermal balance and thermal performance testing, and also during all phases of the mission including on-orbit storage. All transistor collector junction temperatures shall be below 110°C.

10.6.3 ***Uncertainties*** - Predicted temperatures showing compliance with 10.6.1 and 10.6.2 shall account for all uncertainties in calculations, thermal parameters and measurements.

10.7 Structural

10.7.1 ***General Requirements*** - The structural subsystem consists of all spacecraft structural elements, including fixed appendages. A series of tests and analyses shall be conducted to demonstrate the flight hardware is qualified for the expected mission environments, including structural loads, vibroacoustics, sine vibration, mechanical shock, and pressure profiles. The hardware design must also comply with specified verification requirements, such as factors of safety, interface compatibility, structural reliability, workmanship, and associated system safety elements. The spacecraft shall accommodate hard point interfaces for operations such as lifting, rotating, and transporting.

10.7.2 ***Strength Qualification by Test*** - Verification of adequate strength shall be demonstrated by applying a set of loads equal to 1.25 times the limit loads, after which the hardware must be capable of meeting its performance criteria (see section 10.7.2.1 for special beryllium structure requirements). No detrimental permanent deformation shall be allowed to occur as a result of applying the loads, and all alignment requirements shall be met following the test. The strength test must be accompanied by a stress analysis to demonstrate positive margins at ultimate loads equal to 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. In addition, the analysis must show that the maximum allowable loads at the launch vehicle interface points are not exceeded and no excessive deformations occur.

If satisfactory qualification tests have been conducted on a representative model, the strength qualification testing of the protoflight unit may not be necessary, pending NASA review and approval.

10.7.2.1 ***Strength Qualification for Beryllium*** - All Beryllium primary and secondary structural elements shall undergo a strength test to 1.4 times the limit load. In addition: conflict between sections 8.5.3.2 and 10.5.4.1, section 10.5.4.1 shall govern. During each solar array thermal-vacuum cycle test, the outgassing rates of all the flight panels shall be measured with temperature-controlled quartz crystal microbalances (TQCM) to verify that the solar array outgassing requirement is met. If the solar array outgassing requirement is not met, a corrective bake-out phase will be assessed and implemented. For GOES N, obtain at least 12 GOES N solar cells and verify that the average Solar Absorptance is consistent with the established MAT range.

1. When using cross-rolled sheet, the design shall preclude out-of-plane loads and displacements during assembly, testing, or service life.
2. To account for uncertainties in material properties and local stress levels, a design safety factor of 1.6 on ultimate material strength shall be used.
3. Stress analysis shall properly account for the lack of ductility of the material by rigorous treatment of applied loads, boundary conditions, assembly stresses, stress concentrations, thermal cycling, and possible material anisotropy. The stress analysis shall take into account worst-case tolerance conditions.
4. All machined and/or mechanically disturbed surfaces shall be chemically milled to ensure removal of surface damage and residual stresses.
5. All parts shall undergo penetrant inspection for surface cracks and crack-like flaws per ASTM E1417, Standard Practice for Liquid Penetrant Examination.

10.7.3 ***Strength Qualification by Analysis*** - If appropriate development tests are performed to verify the accuracy of the stress model, stringent quality control procedures are invoked to ensure conformance of the structure (materials, fasteners, welds, processes, etc.) to the design, and the structure has well defined load paths, strength qualification may (pending review and approval) be accomplished by a stress analysis demonstrating the hardware has positive margins on yield at loads equal to 1.6 times the limit load, and positive margins on ultimate at loads equal to 2.0 times the limit load. In addition, the analysis shall show that the maximum allowable loads at the launch vehicle interface points are not exceeded and that no excessive deformations occur.

10.7.4 ***Transportation and Handling Loads*** - When transportation and handling loads are not enveloped by the maximum expected flight loads, the transportation and handling loads shall be included in the set of design limit loads.

10.7.5 ***Clearance Verification*** - Analysis shall be conducted to verify dynamic clearances between the spacecraft and launch vehicle and between members within the spacecraft for all significant ground test and flight conditions. Adequate clearances shall be verified assuming worst-case static clearances and quasi-static and dynamic deflections due to 1.4 times the applicable flight limit loads or flight-level ground test levels. Deviations from the 1.4 times factor shall be justified by analysis and with NASA approval.

10.7.6 ***Pressurized Systems*** - The design and verification of all spacecraft pressurized systems shall be in accordance with MIL-STD-1522A, or shall have a degree of compliance commensurate with the safety factor that is provided with procedural controls. The degree of compliance and the procedural control shall be approved by Range Safety.

10.7.7 ***Structural Reliability (Residual Strength Verification)*** - To ensure that adequate residual strength is present (strength remaining after flaws are accounted for), a fracture control program, or a combination of fracture control and specific loads tests, shall be performed on all flight hardware as specified below. If materials are used for structural application that are not listed in Table 1 of MSFC-SPEC-522B, a materials usage agreement (MUA) must be negotiated with GSFC.

The following requirements regarding beryllium, nonmetallic-composite, and metallic-honeycomb structural elements (both primary and secondary), and bonded structural joints shall apply:

1. Beryllium primary and secondary structure: the requirements of section 10.7.2.1 apply.
2. Nonmetallic composite structural elements (including metal matrix): all primary load path structural elements shall be proof tested to 1.25 times the limit load (even if previously qualified on valid prototype hardware). In addition:
    a. The spacecraft contractor shall develop and implement a process control plan to ensure uniformity of processing among test coupons, test articles, and flight hardware.
    b. The spacecraft contractor shall implement a damage control plan to establish procedures and controls to prevent and/or identify nonvisible impact damage which may cause premature failure of composite elements.
3. Metallic honeycomb (both face sheets and core) structural elements: the spacecraft contractor shall implement appropriate process controls and coupon testing to demonstrate the honeycomb structure is acceptable for use as a flight structure. Metallic honeycomb is not considered to be a composite material.
4. Bonded structural joints (either metal-metal or metal-nonmetal):
    a. Every bonded structural joint in a flight article shall be proof tested (by static loads test) to 1.25 times the limit load. Proof test is not required for fail safe joints.
    b. The spacecraft contractor shall develop and implement a process control plan to ensure uniformity of processing among test coupons, test articles, and flight hardware.

Fracture control requirements shall apply to the following elements only:

1. Pressure vessels, dewars, lines, and fittings (per NHB-8071.1).
2. Castings (unless hot isostatically pressed and the flight article is proof tested to 1.25 times the limit load).
3. Weldments.
4. Parts made of materials on Tables II or III of MSFC-SPEC-522B if under sustained tensile stress.
5. Parts made of materials susceptible to cracking during quenching.
6. Nonredundant, mission-critical, preloaded springs loaded to > 25% their ultimate strength.

All glass elements stressed above 10% of their ultimate tensile strength shall also be shown by fracture analysis to satisfy “safe-life” or “fail-safe” conditions, or else shall be subjected to a proof loads test at 1.0 times the limit level.

10.7.8 ***Acceptance Requirements*** **-** All of the structural reliability requirements of section 10.7.7 apply for the acceptance of all flight hardware.

Generally, structural design loads testing is not required for flight structures that have been previously qualified for the current mission as part of a valid prototype or protoflight test. However, the following acceptance/proof load tests are required unless equivalent load-level testing was performed on the actual flight hardware as part of a protoflight test program:

1. Beryllium structure (primary and secondary) shall be proof tested to 1.4 times the limit load.
2. Nonmetallic composites (including metal matrix) structural elements shall be proof tested to 1.25 times the limit load.
3. Bonded structural joints shall be proof tested (by static loads test) to 1.25 times the limit load. Proof testing is not required for fail safe joints.

10.8 Mechanisms

10.8.1 ***Mechanical***

Torque Ratio - The torque ratio shall be determined by test to demonstrate the minimum requirements. The torque ratio is the ratio of the driving or available torque to the required or resistive torque. Numerically, the torque margin is the torque ratio minus one. The torque ratio requirement defined below applies to all mechanical functions, those driven by motors as well as springs, etc. at predicted worst case conditions at end-of-life (EOL). For linear devices, the term “force” shall replace “torque”.

For final design verification, torque ratios shall be verified by testing the qualification unit before and after exposure to qualification level environmental testing. Torque ratios shall also be verified by testing all flight units both before and after exposure to acceptance level environmental testing. Testing shall be performed at the highest level of assembly, throughout the mechanism’s range of travel, under worst-case BOL environmental conditions representing the worst-case combination of maximum and/or minimum predicted (not qualification) temperatures, gradients, positions, acceleration/deceleration of load, voltage, vacuum, etc. The torque ratio T_R is given by:

$$T_R = \frac{T_{avail}}{T_{res}}$$

where, T_{avail} = available torque

T_{res} = resistive torque

Table 10.8.1 indicates the minimum required test verified torque ratios for various classifications of mechanical and electromechanical systems. These classifications are based on the general formula:

$$T_{avail} = 4T_{ru} + 1.25\,T_{rk}$$
$$T_{res} = T_{ru} + T_{rk}$$

where,

T_{avail}	=	total torque available from mechanism or actuator.
T_{res}	=	resistive torque
T_{rk}	=	Torque required to overcome well known resistive torques that will not change over the mission lifetime such as accelerating/decelerating, inertial, or well characterized springs.
T_{ru}	=	Torque required to overcome predicted worst case torques which vary with environmental conditions and operating life such as friction, stiction, or flexing cable assemblies.

The classification of each spacecraft mechanism will be agreed to by the government and the spacecraft contractor so that the appropriate torque ratio requirements are applied and test verified.

10.8.2 ***Electromechanical Stability Margins*** - All electromechanical control systems shall provide the following stability margins, demonstrated by analysis and actual measurement:

Gain Stability $\geq$ 12 dB

Phase Margin $\geq$ 40 degrees

10.9 Flight Software

10.9.1 ***General Requirements*** - All software programs and data for the GOES N-Q spacecraft shall meet the requirements below. The software component of firmware, consisting of computer programs

Table 10.8.1

Torque Ratio Requirements

SYSTEM TYPE	**T_R min**
Systems which are dominated by resistive torques due to inertia, such as momentum and reaction wheels	1.5
Systems which are dominated by resistive torques due to a combination of both inertia and friction, such as large pointing platforms and heavy deployable systems	2.5
Systems which are dominated by resistive torques due to friction, such as deployment mechanisms, solar array drives, cable wraps, and despun platforms	3.5

and data loaded into a class of memory not dynamically modifiable by the computer during processing (e.g., Programmable Read-Only Memories (PROMs), Programmable Logic Arrays, Digital Signal Processors, etc.), shall be specified, designed, developed, configuration controlled, and tested in the same rigorous manner as the flight software.

10.9.2 ***Language and Methodology*** - All software developed for the GOES N-Q spacecraft shall be developed with the ANSI/ISO standard Ada, C, or C^{++} languages and a widely-accepted, industry-standard, formal software design methodology (e.g., structured methods, object-oriented design, object modeling technique, Booch method, Software Cleanroom, etc.). Minimal use of processor-specific assembly language is permitted for certain low-level programs such as interrupt service routines and device drivers with NASA approval.

10.9.3 ***Flight Software Modularity*** - The GOES N-Q flight software shall be written in modular or object-oriented form so that functional units of code can be modified on-orbit with minimal impact to spacecraft operations. The flight software shall be capable of being uploaded in modules, units, segments, or objects which shall be usable immediately after completion of an upload of the modified modules, units, segments, or objects. Activation of the modified modules, units, segments, or objects shall not require completion of an upload of the entire flight software image.

10.9.4 ***Flexibility and Ease of Software Modification*** - The GOES flight software design shall be flexible and table-driven for ease of operation and modification. It shall be rigid in terms of scheduling and prioritization of critical processing tasks to ensure their timely completion. Limits and triggers for anomaly responses shall be readily accessible and changeable by ground command.

10.9.5 ***Responsiveness to Ground Originated Changes*** - The GOES flight software design shall accommodate processing of ground commands, on-orbit revisions to software and telemetry formats, computer self checks, **redundancy** management, and spacecraft mode changes.

10.9.6 ***Software Event Logging in Telemetry*** - The GOES flight software design shall include software event logging in telemetry. These time-tagged messages shall capture all anomalous events, **redundancy** management switching of spacecraft components, and important system performance events.

10.9.7 ***Version Identifiers in Embedded Code*** - All software and firmware shall be implemented with internal identifiers embedded in the executable program or image indicating the version of the currently installed software/firmware to ground operators.

10.9.8 ***Monitoring of Housekeeping and Anomaly Data*** - The flight software shall monitor spacecraft component, subsystem, and instrument housekeeping data, and shall safely configure the spacecraft in the event of an anomaly.

10.9.9 ***Ground Override of Autonomous Anomaly Responses*** - All flight software autonomous functions, automatic safing, or switchover capabilities shall be capable of being overridden by ground command (see section 4.5).

10.9.10 ***Flight Processor Resource Sizing*** - During development, flight processors providing computing resources for spacecraft subsystems shall not be sized for worst case utilization above the capacity utilization goals shown below (measured as a percentage of total available resource capacity):

Table 10.9.10
Flight Processor Resource Utilization Limits

Resource/Phase	S/W PDR	S/W CDR	S/W AR
Memory	40%	50%	60%
CPU	30%	40%	50%
I/O Bandwidth	30%	40%	50%

10.9.11 ***Sizing of Dedicated, Single-function Processors*** - Dedicated, single-function processors, for which maximum loading can be precisely computed, shall be sized such that the following capacity utilization goals (measured as a percentage of the total available resource capacity) are maintained for the worst-case operating modes at launch:

Table 10.9.11
Launch Mode Maximum Flight Processor Utilization Objectives

Resource/Phase	S/W PDR	S/W CDR	S/W AR
Memory	50%	60%	70%
CPU	60%	70%	80%
I/O Bandwidth	60%	70%	80%

10.9.12 ***Software Development and Validation Environment*** **-** A software development and validation environment shall be used for the life cycle management of the GOES spacecraft and associated ground operational software. This environment shall provide the software, tools, and procedures necessary to perform software management, design, development, local configuration management, testing, debugging, integration, and verification, maintenance, and preparation of software images in a format suitable for uplink to the spacecraft.

The software development and validation environment shall include a software test bed that includes both hardware and software to accurately simulate the on-board spacecraft environment in which the flight software will operate. The test bed shall contain engineering models and hardware simulators of necessary flight hardware (such as the spacecraft processor/on-board computer) as well as software models comprising the remainder of the spacecraft, including spacecraft attitude dynamics, sensors, actuators, and the spacecraft environment. The test bed shall provide accurate simulation of software task execution, input/output functions, timing, and dumps.

10.9.13 ***Command Processing*** **-** The flight software shall have the capability of processing real-time, stored (absolute time), and relative-time sequence commands.

10.9.14 ***Memory Location Dump Capability*** **-** The flight software, and associated on-board computer hardware, shall provide the capability to dump any location of on-board memory to the ground upon command.

10.10 On-Board Computer (OBC)

10.10.1 ***OBC Radiation Immunity*** **-** The OBC(s) shall be sufficiently radiation immune/tolerant and redundant to support mission lifetime requirements. The OBC(s) shall be protected against SEUs and other memory and processor errors through the use of design features such as radiation hardened parts, memory error detection and correction (EDAC), periodic software refresh of critical hardware registers, processor and register majority voting, watchdog timers, etc.

10.10.2 ***Uplinked Software Change Implementation*** **-** The OBC(s) shall be capable of implementing uplinked software revisions/modifications/corrections to selected spacecraft processors.

10.10.3 ***On-orbit Reprogrammability without OBC Restart/Reboot*** **-** The OBC(s) shall be reprogrammable on-orbit to allow for new versions of software segments and table values to be loaded from the ground without computer restart.

10.10.4 ***Flight Load Non-volatile Memory*** **-** The OBC shall possess sufficient non-volatile memory to contain the entire flight software load image at launch. The OBC shall be capable of copying the default load image from non-volatile memory to working RAM or operating directly from the image in non-volatile memory upon restart or via ground command.

10.10.5 ***OBC Undervoltage and Transient RAM Event Performance*** **-** The OBC shall have access to hardware switches or non-volatile memory to retain knowledge of the correct spacecraft configuration through spacecraft bus undervoltage events, processor reboots, and processor switchovers.

10.10.6 ***Software Function Availability upon Processor Reset/Reboot*** **-** The following minimum software functions shall be available upon processor reset or reboot, either through bootstrap code in non-volatile memory (PROM, EEPROM, etc.) or through processor hardware discrete commands:

1. Processor RAM loading
2. Processor RAM dumping
3. Initiation of all or an essential portion of the software functions.

10.10.7 ***Deterministic Power-on Configuration*** **-** The OBC shall initialize upon power-up into a known, deterministic configuration.

10.10.8 ***Ground Commandable OBC Reboot/Reinitialization*** **-** The OBC shall provide for reset/reboot/reinitialization of software for recovery from spacecraft anomalies, including software anomalies, by ground command.

10.10.9 ***OBC RAM Content Upload*** - All OBC RAM shall be capable of being uploaded from the ground.

10.10.10 ***Segment, Module, and Object Upload and Dump*** - The OBC memory shall support the upload and dumping of flight software in segments, modules, or objects.

10.10.11 ***Commandability of Redundant Component Configuration*** **-** The operational configuration of OBC redundant components (e.g., processor, bus, memory, non-volatile memory, etc.) shall be commandable and configurable from the ground.

10.10.12 ***Reset/Reboot/Reinitialization Self-test*** **-** The OBC shall be capable of performing an autonomous self-test of the OBC hardware (processor, bus, memory, etc.) upon reset/restart/reboot.

10.10.13 ***OBC Health and Safety Monitoring*** **-** The OBC shall monitor its health and safety during on-orbit operations and shall report the result of its health and safety checks to the ground in spacecraft telemetry.

10.10.14 ***Critical OBC Function Protection*** **-** The OBC shall protect critical attitude control system (ACS) functions from temporary or permanent faults or errors within the processor. This shall include protection against such failure modes as excessive thruster firings, infinite software loops, race conditions, and bus undervoltage conditions.

10.10.15 ***High Speed Test Access Port*** **-** The OBC shall include a built-in test access port, allowing high-speed loading and dumping of processor memory.

10.10.16 ***Memory Location Dump Capability*** - The OBC and associated flight software shall implement the capability to dump any location in memory to the ground upon command.

10.10.17 ***Redundant Component Status*** **-** The OBC shall have the capability to determine the status redundant components before switching spacecraft control to that component and telemetering the status to the ground.

10.10.18 ***Retention of Redundant Component Status*** **-** The OBC and associated flight software shall provide a means by which the spacecraft retains knowledge of the status of redundant components following processor restart, processor failover, RAM memory loss, or bus undervoltage so that the spacecraft avoids switching to previously failed components. This may be implemented in hardware switches, configurable EEPROM, hardened latches (non-volatile for under-voltage time durations less than one second), or another suitable method.

10.10.19 ***OBC Clocks*** **-** The OBC shall provide in telemetry the absolute time codes it uses to process stored commands. The clock(s) shall be resettable via ground command to account for any drift.

10.11 Contamination

The GOES optical and thermal systems are vulnerable to ground based and on-orbit contamination. Degradation of the optical system can occur with significant accumulation of particulate and/or molecular depositions. The following requirements will help preclude potential performance degradation.

10.11.1 ***Facility Requirements*** **-** During assembly, cleaning, system integration and testing, and launch site processing, the spacecraft shall be processed in a Class 100,000 cleanroom per FED-STD-209 (@ 0.5 Fm and 5.0 Fm). A particle counter shall be used to continuously monitor activities where the spacecraft is present, and the Class 100,000 facility shall be recertified to FED-STD-209 every 6 months. Particle counter monitoring for class 100,000 shall be performed at least twice per 8-hour shift. Particulate fallout shall not exceed Level 450 in any 14-day period per MIL-STD-1246. The maximum non-volatile residue (NVR) fallout shall not exceed 0.3 mg/ft^2 in any 1-month period. The maximum allowable total hydrocarbon content shall not exceed 15 ppm (methane equivalent), and the relative humidity shall be maintained between 30 and 60%. Refer to GFE instrument ICDs for additional instrument facility requirements.

All spacecraft shipping and transport containers shall have a maximum air cleanliness of Class 100,000 per FED-STD-209 or Class 10,000 per FED-STD-209 for spacecraft or instrument configurations with exposed sensitive surfaces. For the launch vehicle, the encapsulated spacecraft (i.e., fairing) shall be maintained in a maximum Class 5,000 environment per FED-STD-209 as measured daily. A maximum total hydrocarbon of 15 ppm (methane equivalent) shall be sampled prior to encapsulation and every 5 days thereafter, or as anomalies occur (including severe weather that may compromise environment integrity).

10.11.2 ***Surface Cleanliness Requirements***

10.11.2.1 ***Spacecraft External Requirements*** **-** The external surfaces of the spacecraft shall meet visibly clean highly sensitive (VCHS) criteria. VCHS is define as the absence of particulate and molecular contaminants when viewed at an oblique angle from a distance of 6-18 inches with normal unaided vision (corrective vision lenses acceptable) under $ 100 foot-candle illumination. The NVR requirement for spacecraft external surfaces shall be set at # 2.0 mg/ft^2. For NVR verification, direct flight hardware sampling is preferred. However, representative witness samples are acceptable. The surface cleanliness requirements shall be verified once per month, and before and after the following processing events: integration of instruments, acoustic testing, vibration testing, thermal vacuum testing, after inter-facility transfers, storage (every 6 months), shipment to the launch base, and launch base facility transfers. Spacecraft and/or instrument surfaces shall be bagged or covered during overnight periods or when contamination generating activities occur within the cleanroom facility.

10.11.2.2 ***Spacecraft Contribution to the Instruments*** **-** Refer to the Imager, Sounder, and SXI ICDs for the spacecraft-to-instrument allocations.

Ground processing verification or equivalent shall be performed each time the sensitive surfaces are exposed and during the processing events noted in section 10.11.2.1. For ground processing, a surface that does not lend itself to being sampled directly for NVR may be measured on representative witness samples. Verification of on-orbit operation shall be performed using particle and molecular redistribution computer analyses. On-orbit operations shall include deployables and view factors from other spacecraft surfaces (i.e., solar array, antennas, sun shields, cooler door covers, etc.).

Because of the vibroacoustic environment during launch, contamination redistribution may occur during ascent, stage separation, fairing separation, and orbital insertion. The fairing shall meet level 500A per MIL-STD-1246 for a fairing with no acoustic blanket, with verification via the tapelift sampling method or equivalent. Level 500A is based upon launch and ascent predictions made from the *GOES Contamination Analysis Final Report, Revision A* on the GOES I-J spacecraft orientation in the encapsulated fairing. Provisions shall be made to monitor the fallout environment (either with an in-situ or passive fallout monitor) within the encapsulated fairing using NASA/GSFC document S-415-28 or equivalent. The preceding document is specific to the GOES I-J spacecraft design. In addition, NVR shall be verified at final cleaning and prior to spacecraft encapsulation. All NVR levels greater than 0.5 mg/ft^2 shall be analyzed for contaminant species identification. The fairing shall provide the necessary access for assessing contamination effects on the instruments and coolers.

Particles and gases from the launch vehicle fairing separation pyrotechnics and separation clamp pyrotechnics shall be shielded, directed, or sealed so as to contain contaminants and /or direct them away from the spacecraft.

In the event any surface cleanliness requirements are violated during processing with the spacecraft, the spacecraft contractor shall provide access to the instruments for inspection and cleaning. The spacecraft contractor shall also provide appropriate covers or bags necessary to sustain spacecraft cleanliness requirements for all program phases through final closeouts on the pad at the launch site.

10.11.3 ***Outgassing Requirements*** **-** In order to minimize outgassing of the overall system, all spacecraft and launch vehicle nonmetallic materials shall be screened for total mass loss (TML) and collected volatile condensable material (CVCM) when exposed to a vacuum environment. Polymeric materials shall be screened to meet # 1.0% for TML and # 0.1% for CVCM. Special processing or high temperature vacuum bakeouts are permitted to qualify a high outgassing material, with GSFC concurrence. Refer to NASA RP 1124 for material outgassing data.

All multilayer insulation (MLI) and spacecraft surfaces located near the instrument optical ports and cooler surfaces shall be vacuum baked to meet the following outgassing rate requirement: an outgassing rate of 6.55×10^{-9} g/cm^2-hr as measured with a 15 MHz (sensitivity of 1.56×10^{-9} g/cm^2-Hz) temperature-controlled quartz crystal microbalance (TQCM) averaged over an 8-hour period. The TQCM measurement shall utilize an outgassing test box where the TQCM sensor is located inside a control vented test box. The temperature of the hardware shall be 10°C above the maximum on-orbit predicted temperature, and the TQCM shall collect at -10°C below the on-orbit instrument coldest exposed temperature. All MLI and spacecraft vents shall be directed away from the Imager and Sounder instrument optical ports and cooler surfaces, and away from the SXI optical and thermal control surfaces. Outgassing rates for the solar array shall be established based upon the performance degradation of the SXI optical and thermal surfaces according to on-orbit operational temperatures. All spacecraft panel surfaces with a view to the SXI radiator and aperture plate shall be vacuum baked. MOLEFLUX or equivalent modeling analyses shall be performed using outgassing rates from the solar array and panel surfaces with a view of the SXI to ensure the SXI performance requirements have not been compromised. To minimize deposition on the SXI, the solar array and panel surface, outgassing rates shall be less than 3.12×10^{-7} g/cm^2-hr as measured on both sides of the solar array using a 15 MHz (sensitivity of 1.56×10^{-9} g/cm^2-Hz) TQCM averaged over an 8-hour period. The TQCM FOV shall be completely filled by the solar array. The outgassing rate shall be measured with the solar array at 10°C above the maximum on-orbit operating temperature, and the TQCM at 10°C below the on-orbit operating temperature of the SXI aft radiator plate.

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(Note: Please refer to Appendix A, Deviation and Waiver Requests for Paragraph 10.11.3)

10.11.4 ***Plume Impingement*** - All ascent, transfer orbit, and station keeping thruster firings shall be analyzed to determine if any of the cleanliness requirements in section 10.11.2 are exceeded during the mission life.

10.11.5 ***Purge Requirements*** - Refer to the GFE instrument ICDs and CDRL SDA-3.2.17-01 for instrument purge requirements and launch vehicle requirements.

10.11.6 ***Storage and Transportation*** - During storage and transportation periods, the spacecraft and instruments shall be bagged in ESD protective material meeting surface cleanliness requirements and facility requirements. The hardware or representative witness samples shall be examined/changed out every 6 months during extended storage periods. Any storage containers used during spacecraft storage periods shall not be opened or stored in environments other than those specified in section 10.11.1.

11.0 SUPPORT EQUIPMENT

11.1 Electrical and Mechanical Ground Support Equipment

11.1.1 *Electrical Ground Support Equipment (EGSE) General Requirements*

EGSE comprises all electrical and electronic equipment used for support of the spacecraft during all ground operation phases, and includes: automatic test equipment (ATE), test boxes, breakout boxes, uninterruptible power supplies (UPSs), propulsion control test assemblies, and all other ancillary electrical and electronic equipment used during integration, test, launch support, and flight support. The EGSE devices shall be suitable for their intended purpose, shall be designed to preclude damage to the spacecraft and instruments, and shall not cause a safety hazard to personnel.

11.1.2 ***Mechanical Ground Support Equipment (MGSE) General Requirements*** - MGSE, which includes lifting slings/spreader bars, handling fixtures, shipping containers, etc., shall be provided for mechanical support of the spacecraft during all ground operations.

11.1.2.1 ***Design Factors of Safety*** - Design factors of safety for all MGSE shall be 3.0 on yield and 5.0 on ultimate.

11.1.2.2 ***Proof Test*** - All MGSE shall be proof tested at 2.0 times the maximum expected loads.

11.2 GFE Instrument Test Equipment

11.2.1 ***Spacecraft GSE to GFE GSE Interface*** - The spacecraft GSE shall interface with the GFE instrument GSE used to conduct post-spacecraft integration instrument testing. The interface design shall be jointly developed between the spacecraft and instrument contractors. The spacecraft GSE shall perform the following functions:

1. Receive instrument commands from the instrument GSE, appropriately format them, perform command checking as necessary to protect the spacecraft, and transmit commands to the spacecraft. The spacecraft GSE shall report the receipt, transmission, and execution of instrument commands to the instrument GSE. The spacecraft GSE shall allow for concurrent testing of the GFE instruments, and shall arbitrate between commands sent from the instrument GSE.
2. Transmit baseband (unmodulated) instrument wideband data in real-time to the Imager/Sounder GSE.
3. Transmit baseband (unmodulated) MDL data to the instrument GSE in real-time.
4. Transmit baseband (unmodulated) spacecraft PCM data to the instrument GSE in real-time.
5. Transmit analog test point, accelerometer, and thermocouple data to the instrument GSE.

6. Receive and demodulate instrument RF (modulated) wideband data and transmit it to the Imager/Sounder GSE in real-time.
7. Receive and demodulate instrument RF (modulated) MDL data and transmit it to the instrument GSE in real-time.
8. Transmit instrument-relevant test data, such as chamber telemetry, to the instrument GSE. The content of this data shall be determined jointly by the instrument and spacecraft contractors.
9. Provide a time tag to the instrument GSE.
10. Maintain instrument GSE grounding as defined in the (TBD) ICD.

11.2.2 ***Instrument GSE Accommodation*** - The spacecraft test facilities shall accommodate the GFE instrument test equipment. The spacecraft thermal vacuum chamber shall accommodate infrared targets, space targets, and the associated target controllers, plumbing, and wiring. The chamber shall provide feed-throughs for the instrument target plumbing and wiring. Test equipment accommodations shall be developed jointly by the instrument and spacecraft contractors.

11.2.3 ***GFE Handling Equipment*** - Instrument handling shall comply with the requirements defined in the instrument ICDs. The GFE instruments will be delivered with lifting fixtures. The spacecraft GSE shall interface with these lifting fixtures and/or with the instruments directly to move and orient the instruments as necessary for spacecraft integration.

11.3 Spacecraft Emulator Operations Requirements

NASA, NOAA, and the GFE instrument contractors need to simulate spacecraft functions for a variety of purposes. The spacecraft contractor shall determine the proper combinations of software and hardware to meet the requirements of this section.

11.3.1 ***General*** - The spacecraft emulator shall provide at least one hardware OBC (not necessarily flight rated) to assist in the development, testing, and generation of flight software patches. The emulator's OBC is not required to operate in other than real-time. Realistic modeling shall be provided for all spacecraft subsystems. Thermal modeling shall include both orbit raising and on-orbit configurations. Mechanical modeling of deployments shall include sensors for measuring time-variant inertia, center of pressure, and center of mass. Propulsion subsystem modeling shall include tank and line pressure variations. The emulator shall provide space environment modeling, including accurate representations of the motion and properties of the Moon, Earth, Sun, and stars. No operations which can be performed on the spacecraft shall be precluded from being performed on the emulator. The spacecraft emulator shall interface with the GFE instrument simulators and the actual instruments in exactly the same way, such that the component being used (GFE simulator or actual instrument) is completely transparent to the emulator user. This includes utilization of all databases, commands and procedures, although setup steps may be different. The spacecraft emulator shall perform the following functions:

11.3.1.1 ***Data Stream Generator*** - The data stream formats of the emulator and the spacecraft shall be identical. The emulator shall provide correctly formatted streams for any type of data originated on the actual spacecraft, including housekeeping telemetry, MDL telemetry, and raw wideband telemetry. For this purpose, no dynamic data is required.

11.3.1.2 ***Command Processor*** - The emulator shall accept all secure and clear mode commands, OBC uplinks, and dump requests, and shall respond by changing the digital telemetry as commanded. The emulator shall provide realistic modeling of command receipt processing and execution, including command delays. The emulator shall provide realistic modeling of the housekeeping telemetry stream, including accurate timing and formatting.

11.3.1.3 ***Imager/Sounder INR Test Interface*** - The emulator shall provide all signals that the spacecraft INR subsystem provides to the Imager and Sounder instruments. The emulator shall generate realistic E-W and N-S IMC signals based on data from the Imager and Sounder (i.e., such as scan direction and frame size) and data derived from other spacecraft subsystem models (i.e., such as attitude and orbit information). The emulator interface to the Imager and Sounder shall be identical to the actual spacecraft INR interface to the Imager and Sounder. The emulator shall safely interface with flight Imager and Sounder instruments.

11.3.1.4 ***SXI Test Interface*** - The emulator interface to the SXI shall be identical to the actual spacecraft interface to the SXI. The emulator shall safely interface with flight SXI instrument.

11.3.1.5 ***Training Tool*** - The emulator shall respond to all commands in the same manner as the spacecraft, including changing all analog telemetry based on orbit and attitude models, space environment models, telemetry and command link models, realistic thermal effects, and full operation of all instruments (images not required). The emulator shall have the ability to start from any date or time, to be paused indefinitely, to rapidly advance or back up to any point in the orbit, and to store its state electronically and reload the stored state. The emulator shall provide a means of producing contingencies to spacecraft operation. These shall include abrupt failure, degraded performance, or erratic behavior of any sensor, actuator, or mechanism on the spacecraft. The telemetry generation shall also allow the insertion of values into the telemetry stream that are not overridden by software models. Contingencies shall be emulated for all spacecraft/instrument modes of operation and all subsystems operating during those modes.

11.3.1.6 ***Engineering Tool*** - The emulator shall function as a test bed prior to operations on the actual spacecraft. This includes verification of OBC loads, flight software patches, maneuver planning, ground system validation, and changes to operational spacecraft procedures.

11.3.1.7 ***Analysis Tool*** - The emulator shall include the capability to run in an accelerated mode to allow the performance of long duration spacecraft simulations. A portion of the OBC functionality shall be provided in software for this purpose, precluding the need for OBC hardware to run at other than real time.

11.3.2 ***Operations Network Compatibility*** - The spacecraft emulator shall be compatible with both the launch and on-orbit operations network. The spacecraft emulator shall be capable of transmitting downlink data to the SOCC and CDASs in a form that is compatible with the ground system ingest equipment. The emulator shall be capable of processing uplink data received from the SOCC and CDASs via the operations network. The emulator shall appear to the SOCC to be sending data from any ground station site used in the actual mission.

11.3.3 ***Remote System Accessibility*** - Each spacecraft emulators shall be accessible from either the government operations center or the spacecraft contractor's facility.

11.3.4 ***Performance Requirements*** - The spacecraft emulator shall possess at least 15% margin on all resources (CPU, storage, etc.) at delivery to accommodate future enhancements (e.g., emulator upgrades, emulations of spacecraft/instrument upgrades).

11.3.4.1 ***Reliability*** - The emulator shall have a Mean Time Between Failures (MTBF) of no less than six months.

11.3.4.2 ***Simultaneous Spacecraft Simulations*** - The spacecraft emulator shall have the capability to perform simulations of up to three spacecraft simultaneously. Of the three simulated spacecraft, one is required to have all high-fidelity components available; the other two simulated spacecraft need only produce static data streams.

11.3.5 ***Functional Requirements*** - The emulator shall maximize the use of COTS hardware and software, and shall have a GUI as the primary interface for the user. The emulator shall be reconfigurable from one spacecraft mode to another in less than 15 minutes. The emulator shall be fully compatible with the NOAA ground system.

11.4 INR Performance Evaluation System

The INR performance evaluation system shall accurately model/simulate all known INR pointing error sources. As a minimum, the performance evaluation system shall provide the following evaluation and diagnostic capabilities:

1. Perform parametric studies of INR performance.
2. Evaluate the impact of system and subsystem performance.
3 Substitute one or more files derived from real data in lieu of simulated data as an additional means of evaluating the impact of system and subsystem performance changes.

4. Support real-time and near real-time operations for spacecraft level test support at the spacecraft contractor’s plant and in support of end-to-end tests.
5. Use parts of the simulator software to verify and validate operational compatibility of the INR portions of the ground system with the spacecraft.

12.0 ACRONYM LIST

Refer to CODE 415 GOES Acronyms and Abbreviations List

13.0 DEFINITIONS

Adjacent Angular Pixel Separation (Instrument Specification)

The East-West (E-W) angular separation between adjacent pixels in the visible channel is specified to be within 28 μradians +2/-0%.

The North-South (N-S) angular separation between adjacent lines in the same scan line for the visible channel is specified to be within 28*M ±7 μradians, where M is the separation between detectors (an integer); there are 8 visible detectors per scan line.

The North-South (N-S) angular separation between adjacent scan lines in the visible channel is specified to be within 13 μradians.

Adjacent Channel Interference

Mission signal frequency products of a transmission channel higher or lower in center frequency overlapping into the transmission channel-under-test's usable bandwidth.

AM/PM Conversion

AM/PM conversion is characterized by the peak absolute value of the derivative of the amplifier output phase relative to the input over the dynamic range of the transponder or the operating range of the transmitter.

Centroid

Centroid is defined as the intersection of the centers of the half-power points (50% peak amplitude levels) in the N-S and E-W axes.

Component

A functional subdivision of a subsystem and generally a self-contained combination of items performing a function necessary for the subsystem's operation.

Coregistration

Coregistration pertains to the alignment between two channels in the same instrument. Of particular interest are the alignments between the Imager visible channel and IR channels 2 and 4, and the Sounder star sensing channel and channel 8.

Dynamic Interaction

The magnitude of spacecraft modal frequencies measured at the instrument to determine the effect on the INR performance.

Dynamic Gridding

Selection of the dynamic gridding mode of operation will result in being able to determine the location of each pixel in an image or sounding from the data in the retransmitted (GVAR) data. In the dynamic gridding mode the WIFR and FFR requirements are not applicable.

Dynamic Range

The input power range available at the spacecraft receive input port assuming a 0 dB isotropic (dBi) antenna over which the channel is operational (i.e., meets specification).

Effective Isotropic Radiated Power (EIRP)

The transmitted signal power under modulated conditions which would have to be uniformly radiated over 4ð steradians to produce the signal actually received at the input of the receive antenna with the specified polarization and polarization orientation. For the DCPR and SAR channels, the concept of signal power is modified to be the total signal plus noise output power.

Fixed Grid Resulting in Fixed Earth Location Image

When scanning the same Earth sector, a fixed grid image results in corresponding pixels in successive images observing the same (fixed) Earth location, within the navigation pointing capabilities of the system. As a result, the use of fixed (preassigned) grids may be used when scanning the same Earth sectors.

"Fixed grid" in this specification shall mean that images from the same imager channel specified to observe the same geographic area on the Earth shall be corrected for orbit and attitude effects and, therefore, shall be able to use the same Earth-located coordinates.

Frame to Frame Registration (FFR)

FFR pertains to the variation in the locations of two corresponding pixels from two images of the same area of the Earth.

Frequency Stability

The peak instantaneous carrier frequency deviation from the nominal carrier frequency, normalized to the nominal carrier frequency as observed over the specified time interval of interest.

Full Disk

A full-disk GOES image as viewed from the instrument spans the Earth from the satellite nadir to ±8.7° N-S and either +8.7 to -10.4° or +10.4 to -8.7° E-W depending on the Earth side that the space clamp is taken.

Gridding

Gridding of Imager data is the assignment of the locations of selected latitude, national and geographic boundaries to pixels in the image products.

Housekeeping Periods (HK)

HK periods refer to the impact to normal operations required to maintain spacecraft operations or INR operations. Examples are thruster firings to unload momentum wheels, trim tab positioning, if required, and/or special commanding requirements for INR, if required.

Image Navigation and Registration (INR)

INR pertains to the monitoring and control of satellite functions and disturbances (internal and external) so as to be able to determine the location (navigation) of each pixel in latitude and longitude; when desired the INR knowledge can be used to control successive pixels taken at the same instrument elevation and azimuth scan angles to observe the same Earth location, within the pointing capability of the system: referred to as Fixed Earth Location (fixed grid).

Image Motion Compensation (IMC)

In the GOES I/M series, IMC refers to the process of repointing the instrument mirrors to correct/compensate for known orbit/attitude pointing errors; when IMC is active this results in the same Earth area being imaged (i.e., appearing) in the same pixel for a given image or sounding.

In the GOES NO/PQ series, IMC refers to the signal lines into the Imager and Sounder that will cause the mirrors to be repointed up to ±8000 µRadians in the N-S or E-W scan directions in the IMC high range, or ±4000 µradians in the N-S or E-W scan directions in the IMC low range.

Image/Sounding area

Image or Sounding area is defined as the composite of the contiguous set of pixels constituting the area.

Low/High Frequencies

Low frequencies are less than or equal to the respective roll, pitch and yaw (if applicable) attitude control system bandwidths; and high frequencies are greater than the respective attitude control system bandwidths.

Navigation

Navigation pertains to the location of the centroids of pixels relative to the instrument scan angle; for convenience this can be related to latitude and longitude on the Earth.

Normal Operational period

Normal operational periods are the intervals between stationkeeping operational periods.

Pixel/Sounding Sample

A pixel or sounding sample is a picture or sounding element which is obtained by sampling the signal generated by the Imager or Sounder as it scans a line or sounding..

Remapping

Remapping is the movement in E-W and/or N-S of visible or IR pixels relative to their current location with respect to pixels in another channel to effect a change in coregistration.

Resampling

Resampling is the determination of a new visible or IR pixel value (i.e., at a point not previously observed) by mathematical means (e.g., cubic convolution) based on measured/observed pixel values near the new pixel location.

Scan Lines

A scan line is the image or sounding that is obtained from one sweep of the instrument from E-W or W-E: For the Imager, the result is eight visible lines and 2 lines each from IR channels 2, 4 and 5, for a total N-S swath of 224 µradians. For the Sounder, the N-S swath is 1120 µradians.

Spurious Emissions

A spurious emission is defined as any undesired frequency or frequencies (either in-band or out-of-band) not part of the theoretical spectrum of the mission signals at the spacecraft input. Examples of spurious emissions are EMI, images, frequency conversion products, spurious phase and amplitude products, harmonics, and passive intermodulation products. A harmonic emission is a spurious emission at frequencies which are multiples of those contained in the mission signal. This section does not apply to active intermodulation products due to the transmission of multiple signals through a common transmission channel.

Stationkeeping (SK) Operation

A stationkeeping operation refers to the adjustment of a satellite's orbit in order to maintain its geosynchronous position (station). Stationkeeping operations consist of the stationkeeping maneuver period and a stationkeeping recovery period.

Subsystem

A functional subdivision of a spacecraft consisting of two or more components.

Three Sigma (3ó) Value

The 3ó value (where ó is the root mean square value about the mean) is defined as the value for which 99.7% of a measured, calculated and/or determined set of values will be within the specified range. This is equivalent to the 3ó value for a Gaussian distribution.

Transmission Channel

A signal path within the spacecraft, of a specified frequency band, performing receiving, transmitting or relaying of RF signals.

Transponder

Communications hardware, minus the antennas, used to establish a particular end-to-end signal, or transmission channel, between a pair of spacecraft receive and transmit ports in a particular bandwidth or channel. It is expected that certain communication subsystem components will be associated with more than one transponder (e.g., receivers).

Within Frame Registration (WIFR)

WIFR pertains to the variation in separation of two pixels within the same image from the expected separation determined from the adjacent angular pixel separation.